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TS-2A AIRTANKER EVALUATION.

PHASE II.

(9) FINAL REPORT,

(14) USAAEFA-77-07

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EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The United States Army Aviation Engineering Flight Activity performed a limited handling qualities and performance evaluation of the TS-2A airplane modified with a retardant tank system for use in the United States Forest Service airtanker mission. The evaluation was conducted in two phases: Phase I, prior to installation of the liquid dispensing tanks, and Phase II, following installation of the tank modification. Phase I was conducted in 1972 by USAAEFA. The Phase II

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evaluation was conducted from 3 October through 6 December 1977 at Edwards Air Force Base (elevation 2302 feet) and South Lake Tahoe Airport, California (elevation 6262 feet). Thirty-three flights were accomplished for a total time of 50 hours, of which 36.5 hours were productive. The tanker equipped TS-2A airplane is marginal, but satisfactory for the airtanker mission. With few exceptions, the information contained in the NATOPS flight manual as modified by the FAA supplement is correct and appropriate. Recommended changes to the NATOPS manual and FAA supplement include: (1) dynamic minimum-control airspeed of 100 knots indicated airspeed; (2) rotation airspeed of 100 knots indicated airspeed for heavy gross weight operations; (3) lift-off airspeed of 105 knots indicated airspeed for heavy gross weight operations; and (4) the inclusion of accelerate/stop data generated during this test, plus a NOTE relative to abort takeoff emergency procedures. The pitch-up tendency during load drop is a potentially dangerous problem; however, using techniques developed during the evaluation can reduce and/or compensate for the tendency. The techniques include (1) the use of two one-half salvos in rapid sequence, (2) the use of 2/3 flaps during the drop, and (3) the use of the anticipate method whereby the pitch control is placed to the full forward position immediately after the load drop, before the pitch-up has developed. Modifications should be made to preclude retardant contamination of the nose wheel well, and following the modification, extended landing gear should be used during the drop sequence. An investigation should be conducted to determine if a Navy recommended control system modification has been incorporated on the tanker fleet aircraft, with the recommendation that the modification be accomplished as soon as practical if not previously done. This modification may require a change in the anticipate technique. Consideration should be given to raising the FAA limit load factor for retardant drop operations.

DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
P O BOX 209, ST. LOUIS, MO 63166

DRDAV-EQ 28 SEP 1978

SUBJECT: TS-2A Airtanker Evaluation Phase II

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1. The purpose of this letter is to establish the Directorate for Development and Engineering position on the subject report with specific comments on the conclusions and recommendations. In general, we do not believe an increase to the FAA approved limit load factor at a given gross weight for retardant drop operations is required or desired. The load factor limit of 3.0g by the US Navy corresponds to a 23,000 pound gross weight and 2.5g by the FAA corresponds to a 27,000 pound gross weight. These values are equivalent to a 3.6g value at a 19,000 pound gross weight. The 19,000 pound gross weight is representative of the post drop gross weight and is equivalent to a 3.6g structural capability. As evidenced by the test results, a 3.6g capability is adequate. What is required is that the FAA provide the allowable limit load factors for gross weights of 27,000 pounds and 19,000 pounds for flap settings of 0, 1/3, 2/3 and full down.

2. Other specific comments by paragraph are as follows:

a. Paragraph 39 - This paragraph, as written, is misleading in that it implies allowable structural limits are routinely exceeded due to the pitch-up. The TS-2A is good for a load factor of 3.6g at 19,000 pounds with flap up or 1/3 down. During the tests, 3.6g was not exceeded.

b. Paragraph 43 - The next to last sentence identifies the FAA 130 KIAS as being marginal but satisfactory, yet no recommendation is made to at least use a value of 130 KCAS which the test data substantiates as being okay. In fact, the data indicates, especially with the recommended drop technique, that a value of 135 ICAS would be adequate.

c. Paragraph 48 - The basis for the 19,000 pound limit load factor values for flaps up, flaps 1/3 and flaps down was that of keeping the product of limit load factor times gross weight equal to that corresponding to the US Navy and FAA values. The limit load factor for flaps 2/3 was that which conservatively would result in the same flap hinge moments as that which would occur with flaps full down at a 23,000 pound gross weight and 2.0g previously and currently approved for US Navy operations.

DRDAV-EQ 28 SEP 1978
SUBJECT: TS-2A Airtanker Evaluation Phase II

The load factor limit for 19,000 pounds was used for the drop test since the aircraft gross weight was below 19,000 pounds before the pitch-up occurred. The use of 2/3 flaps for the drop sequence is the recommended flap setting for the mission requirements. The FAA STC should be revised to state approved limit load factor values for appropriate gross weights (27,000 and 19,000 pounds) at flap settings of 0, 1/3, 2/3 and full down.

d. Paragraph 56 - Subject to and contingent upon FAA approval, the following limitations should be observed:

(1) Flight Load Accelerations (Load Factor)

(a) Symmetrical Pullout	27,000 lb GW	19,000 lb GW
Flaps up	2.5	3.6
Flaps 1/3	2.5	3.6
Flaps 2/3	2.5	3.6
Flaps full down	1.7	2.5

(b) Unsymmetrical (Rolling Pullout) - allowable load factor is 80% of the symmetrical values.

(2) Maximum retardant drop speed - 135 KCAS.

3. This Directorate agrees with all other recommendations of the report and considers the investigation to determine if the Navy's recommended flight control system modification has been performed on the tanker modified T3-2A aircraft to be very important.

FOR THE COMMANDER:

for Charles Crawford

WALTER A. RATCLIFF
Colonel, GS
Director of Development
and Engineering

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INTRODUCTION

BACKGROUND

1. The TS-2A aircraft was produced by Grumman Aircraft Engineering Corporation for United States Navy antisubmarine warfare operations. Surplus TS-2A aircraft have been released by the Navy to the United States Forest Service (USFS) for modification and use in the airtanker mission. In 1972 the USFS requested the United States Army Aviation Systems Command (AVSCOM), since redesignated the United States Army Aviation Research and Development Command (AVRADCOM), to evaluate the TS-2A aircraft for the airtanker mission. The evaluation was to be conducted in two phases: Phase I, prior to installation of the liquid-dispensing tanks, and Phase II, following installation of the tank modification. Phase I tests were conducted in 1972 (ref 1, app A) by the United States Army Aviation Systems Test Activity, since redesignated the United States Army Aviation Engineering Flight Activity (USAAEFA). In February 1977, USAAEFA was directed by AVSCOM to accomplish Phase II of the evaluation (ref 2).

TEST OBJECTIVES

2. The test objectives of the Phase II evaluation were to determine a safe airtanker flight and load drop envelope for the USFS.

DESCRIPTION

3. The test aircraft was a modified Navy TS-2A, FAA Aircraft Registry Identification No. N404DF, incorporating an 800-gallon storage tank mounted in the fuselage for carrying fire retardant liquids. This tank extended approximately 4 inches below the bottom of the fuselage and had four compartments which allowed selective delivery of the retardant. The basic aircraft, a high-wing twin engine monoplane powered by two Wright R1820-82A engines, was originally designed for short-field takeoff and landing.. The aircraft is certified for a maximum design gross weight of 27,000 pounds, with a maximum landing weight of 24,500 pounds. The test aircraft was typical of other tanker-converted TS-2A aircraft in use by the Forest Service with the exception that the battery was moved from the right engine nacelle to the tail. The battery transfer resulted in an aft center of gravity (cg) shift of 1.28 inches (1.3 percent mean aerodynamic chord (MAC)) and caused only a very slight change in handling qualities. A detailed description of the aircraft and tank installation is contained in appendix B.

4. The mission of the TS-2A aircraft as used in the airtanker role is to operate from improved and unimproved airfields located throughout the fire areas, and to drop retardant accurately in the path of the fire as directed by an air controller.

TEST SCOPE

5. The USFS TS-2A airtanker was evaluated at Edwards Air Force Base (elevation 2302 feet) and South Lake Tahoe Airport (elevation 6262 feet), California, from 3 October through 6 December 1977. During the program 33 flights were flown for a total of 50 flight hours of which 36.5 were productive test hours. A limited performance, handling qualities, and fire retardant delivery technique evaluation was conducted. Aircraft stall characteristics and emergency procedures were also investigated. The aircraft configurations evaluated are shown in table 1 and the general test conditions are shown in table 2. Flight restrictions and operating limitations were dictated by the NATOPS flight manual (ref 3, app A, and the airworthiness release (ref 4).

TEST METHODOLOGY

6. Established flight test techniques and data reduction procedures were used during this test program (refs 5 through 8, app A). The test methods are described briefly in the Results and Discussion section of this report. Flight test data were hand-recorded from test instrumentation on the pilot and copilot instrument panels and automatically recorded on magnetic tape. A detailed list of the test instrumentation is contained in appendix C. Weight and balance and data reduction techniques are contained in appendix D.

Table 1. Aircraft Configurations.

Configuration	Gear Position	Flap Setting[1]	Power Setting (in. Hg MAP)[2]	Propeller Speed (rpm)
Takeoff (TO)	Down	1/3	TOP[3]	2800
Climb (CL)	Up	Zero	45 (single engine)	2500
		Zero	35 (dual engine)	2300
Level flight (LF)	Up	Zero	PLF[4]	2300
Landing (L)	Down	Full	Idle	Maximum
Descent (D)	Up	Zero to full	PD[5]	2500
Drop (DR)	Up/Down	Zero to full	PLF[4]/PD[5]	2400

[1]Flap settings: 0, 1/3, 2/3, full flaps (38.5°).
[2]In. Hg MAP: Inches of mercury manifold pressure.
[3]TOP: Takeoff power: - 49 in. Hg MAP used at Edwards AFB, 41 in. Hg MAP used at Lake Tahoe Airport.
[4]PLF: Power required to maintain level flight.
[5]PD Power required to maintain 500-ft/min or 1000-ft/min rate of descent.

Table 2. General Test Conditions

Test Conditions[1]	Indicated Airspeed (kt)	Pressure Altitude (ft)	Takeoff Gross Weight (lb)	Configuration
Airspeed calibration	90 to V_H[2]	6000	19,000	LF, D, CL
			26,000	
Trim control characteristics	80 to 200	6000	19,000	LF, DR, CL
			26,000	
Static longitudinal stability	120 ±20	6000	19,000	LF
			26,000	
Static lateral-directional stability[3]	120	6000	19,000	LF
			26,000	
Lateral control characteristics	120	6000	19,000	LF
			26,000	
Single-engine minimum control speed	V_S[4]	10,000	19,000	LF
			26,000	TO
Dual-engine stalls[5]	V_S	10,000	19,000	LF, DR
			26,000	
Maneuvering stability	120	6000	19,000	LF
	140		26,000	
Climb performance	100 to 140	6000 ±500	19,000	CL
			26,000	
Takeoff performance	75 to 105[6]	2300	19,000	TO
		6200	26,000	
Landing performance	85 to 105[7]	2300	19,000	L
		6200		
Accelerate/stop performance	60 to 100[8]	2300	19,000	TO
		6200	26,000	
Load drop envelope[9]	110 to 140	6000	26,000	DR
		2500		

[1]All tests conducted at forward cg: Fuselage station (FS) 215 to 216 inches.
[2]V_H: Maximum airspeed for level flight.
[3]Tests conducted with rudder assist OFF.
[4]V_S: Stall airspeed.
[5]Accelerated stalls were performed with power ON only (2400 rpm, 20 in.Hg MAP). Unaccelerated stalls were performed with power ON and power OFF (2400 rpm, 20 in.Hg MAP) or (2400 rpm, IDLE).
[6]Rotation airspeed.
[7]Approach airspeed: Held to a height of 50 feet above ground level (AGL).
[8]Abort takeoff airspeed: Airspeed at which power was reduced to idle and brakes applied.
[9]Retardant drop conducted at 2500 feet. Water drops at 6000 feet.

RESULTS AND DISCUSSION

GENERAL

7. The TS-2A airtanker was evaluated from 3 November through 6 December 1977 at Edwards AFB and South Lake Tahoe Airport, California. The purpose of this evaluation was to determine a safe operating and load drop envelope.

8. For most performance and handling qualities characteristics, the NATOPS flight manual and FAA flight manual supplement (ref 9, app A) were determined to be correct and appropriate for the airtanker. Recommended changes to the NATOPS flight manual and the FAA supplement included noting that the dynamic minimum-control airspeed (V_{MC}) is 100 knots indicated airspeed (KIAS), rotation airspeed is 100 KIAS, and lift-off airspeed is 105 KIAS during heavy gross weight operations. The accelerate/stop performance information generated during this test should be included in the supplement to the flight manual with a NOTE relative to abort takeoff emergency procedures.

9. Numerous techniques were evaluated for the load drop, and it was determined that the use of 2/3 flaps, jettisoning two rapid half-loads rather than a full salvo, and use of the anticipate technique (placing the control yoke to the full forward position immediately after jettisoning the load and before pitch-up begins) combined to produce the least pitch-up and are recommended for all drops. The use of extended landing gear during the drop sequence significantly improved airspeed control and caused a marked decrease in the pitch-up tendency; however, minor modification will be required to preclude retardant contamination of the nose wheel well before this technique can be recommended. An investigation should be conducted to determine if a Navy recommended longitudinal control system modification has been incorporated on the tanker fleet aircraft, with the recommendation that the modification be accomplished as soon as practical if not previously done. This modification may require a change in the anticipate technique. An analysis of the manufacturer's airframe structural limitations should be conducted to determine if the aircraft limit load factor could be safely raised for retardant drop operations.

PERFORMANCE

Takeoff Performance

10. The takeoff performance of the TS-2A aircraft was evaluated at 19,000- and 26,000-pound takeoff gross weights at Edwards AFB and South Lake Tahoe Airport under the general conditions listed in table 2. The aircraft was accelerated by applying takeoff power at brake release. Rotation airspeed was varied on successive runs in 5-knot increments from 75 to 105 KIAS. Rotation was initiated by moderate aft movement of the control yoke and the landing gear was raised as

soon as safely airborne. A minimum airspeed of 105 KIAS was obtained as soon as practical after takeoff as recommended by the NATOPS flight manual and FAA supplement. The data reduction methods used are presented in appendix D, and the test results are presented in figures 1 and 2, appendix E.

11. During the light gross weight tests at both altitudes, the aircraft rotated crisply for rotation airspeeds above 85 KIAS and lift-off airspeed was essentially the same as rotation airspeed. No unusual tendencies were noted. Due to V_{MC} considerations, the Navy recommends and the Forest Service is currently using a rotation airspeed of 93 KIAS and a lift-off airspeed of 98 to 100 KIAS. For light gross weights, this procedure should be continued.

12. During the heavy gross weight tests at both altitudes for airspeeds at and below 95 KIAS, aft control input to initiate rotation would result in nose wheel lift-off and cause the tail wheel to drag; however, lift-off did not occur until approximately 100 KIAS. Rotation at 100 KIAS was relatively crisp with good control harmony in all axes, whereas rotation airspeeds of 90 KIAS and below produced a very sluggish, delayed rotation and relatively poor control harmony. Figure 2, appendix E, shows essentially the same lift-off airspeed (94 to 105 KIAS) for all rotation airspeeds. Dynamic V_{MC} is discussed in paragraph 34 and was determined to be 100 KIAS. A rotation airspeed of 100 KIAS and a lift-off airspeed of 105 KIAS should be used and changes should be made to the FAA flight supplement for heavy gross weight operation. The higher rotation airspeed is safer from V_{MC} considerations and is qualitatively preferable due to improved control harmony.

Climb Performance

13. Climb performance was evaluated under the general conditions listed in table 2. Heavy and light gross weight as well as dual- and single-engine climbs were evaluated. Climb performance was measured by stabilizing the aircraft in a climb at constant airspeed while holding handbook recommended propeller speed (rpm) and manifold pressure. Each climb was repeated on reciprocal headings to minimize wind shear effects. Airspeed was varied in 5- to 10-knot increments from 100 to 140 knots calibrated airspeed (KCAS). The data reduction methods used are presented in appendix D. The results of this test are presented as figures 3 through 5, appendix E.

14. Figure 3 presents dual-engine heavy gross weight climb performance at 7680 feet density altitude (H_D). The maximum rate of climb occurred at 126 KCAS, which is 6 knots faster than the current recommended NATOPS and FAA supplement climb airspeed of 120 KCAS (122 KIAS). Since there is a negligible difference in rate of climb between 120 and 126 KCAS, the NATOPS and FAA climb schedule should be followed. Figure 4 presents dual-engine light gross weight climb performance and indicates a maximum rate of climb occurring at an airspeed below 100 KCAS. However, at the light gross weight, climb airspeeds below about 110 KIAS required uncomfortably high pitch attitudes and reduced the field of view. Figure 5 shows that single-engine climbs at heavy gross weight

also reach a maximum rate of climb at or below 100 KCAS. Performance in excess of 475 ft/min is available in both cases at 120 KCAS, which is the NATOPS and FAA supplement recommended climb airspeed. The increased airspeed affords a safety margin above the dynamic V_{MC} of 100 KIAS (para 34). The recommended climb airspeed is 120 KCAS (122 KIAS) for all conditions. The best angle of climb airspeed is 105 KIAS at or below 19,000 pounds gross weight and 122 KIAS at a gross weight of 26,000 pounds.

Accelerate/Stop Performance

15. The accelerate/stop performance of the TS-2A aircraft was evaluated at light (19,000 pounds) and heavy (26,000 pounds) gross weights at the high and low altitude test sites at the conditions listed in table 2. The aircraft was accelerated by applying takeoff power at brake release and maintaining it to a predetermined airspeed, at which time the throttles were retarded to idle and maximum braking applied without wheel lockup. This procedure was repeated throughout a representative airspeed range (60 to 90 KIAS at Edwards AFB and 70 to 100 KIAS at South Lake Tahoe). The water load was jettisoned once at each test site to determine any advantages to be gained from releasing the load and to evaluate the abort takeoff emergency procedures. On-board data systems as well as external measuring equipment were used. A description of the data analysis method used is presented in appendix D. The results of this test are presented as figures 6 through 11, appendix E.

16. Figures 6 and 7, appendix E, present the total runway distance required to accelerate and stop the aircraft for various gross weight and runway altitude combinations. With a heavy gross weight and at an elevation of 6000 feet density altitude, a 5000-foot runway would be required to safely accelerate to and stop from the recommended rotation airspeed of 100 KIAS. Lower density altitudes, lower gross weight, and lower airspeeds reduced the runway length requirement for an aborted takeoff.

17. Figures 8 and 9, appendix E, present the accelerate/stop performance for light and heavy gross weights at the same altitude. The data at the low altitude test site (fig. 9) indicate that by following the recommended abort takeoff procedures (intermittent braking beginning at 85 KIAS followed by load drop at 70 KIAS and then maximum braking) approximately 400 feet less braking distance was required than if the load had been retained. The data at the high altitude test site (fig. 8) show a maximum benefit of approximately 200 feet of runway saved by dropping the load rather than retaining it (the difference between the heavy gross weight and light gross weight braking distance at the South Lake Tahoe elevation). The current emergency procedure requires a number of pilot decisions to be made under stress with a high possibility for error. Releasing the load at airspeeds higher than the recommended 70 KIAS was evaluated at the South Lake Tahoe Airport by dumping the load at 90 KIAS. The aircraft, with brakes applied, decelerated faster than the water load, allowing water to soak the brakes and also caused hydroplaning (this data point is not presented). No pitch-up tendency was observed; however, the tendency to hydroplane could easily lead to uncontrolled

skids and blown tires. The decision of whether to retain or jettison the load in the event of an aborted takeoff should be based on judgment and precise knowledge of a number of factors, including runway length, runway condition, overrun, density altitude, and aircraft condition and configuration. The decision on abort procedures should be made prior to starting the takeoff. The accelerate/stop performance information contained in figures 6 through 11 should be included in the Flight Supplement along with the following NOTE:

NOTE

> The decision of whether to retain or jettison the load in the event of an aborted takeoff should be based on judgment and precise knowledge of a number of factors, including runway length, runway condition and, overrun, density altitude, and aircraft condition and configuration. The decision on abort procedures should be made prior to starting the takeoff. Although jettisioning the load may result in less runway distance used for braking (up to about 400 feet at lower density altitudes), jettisoning the load also increases the possibility of fouling the gear with brake lock-up and hydroplaning.

<u>Landing Performance</u>

18. The landing performance test of the TS-2A airplane was conducted at 19,000 pounds gross weight. The test was accomplished by flying a normal approach angle (approximately 3.5 degrees) in 5-knot increments at airspeeds from 75 to 105 KIAS. Airspeed was held constant down to 50 feet AGL, at which time attitude and power changes were made as necessary to accomplish a normal landing. The tests were conducted under the general conditions listed in table 2. A description of the data analysis methods is presented in appendix D.

19. Minimum safe airspeeds using the normal approach technique were determined to be 75 KIAS at Edwards AFB, and 80 KIAS at South Lake Tahoe. At these airspeeds, barely enough forward velocity was available to cushion the landing without increasing power. Approaches at 85 and 90 KIAS (the NATOPS recommended approach speed) were comfortable and safe at both locations. The approach at 90 KIAS was additionally enhanced because the higher airspeed afforded a larger safety margin in the event of an engine failure while still affording excellent glide path control with either power or attitude changes. Approach speeds of 95 through 105 KIAS resulted in excessive landing distances even with both engines at idle power.

20. The 90-knot approach speed to the 50-foot height resulted in touchdown speeds of 87 knots at both Edwards AFB and South Lake Tahoe. Total landing distance was 1790 feet at Edwards AFB and 1900 feet at South Lake Tahoe at 19,000 pounds gross weight. Data from interpolation of the NATOPS landing distance chart agree well with the test results. The NATOPS recommended approach speed of 90 KIAS should be used.

HANDLING QUALITIES

Control Positions in Trimmed Forward Flight

21. Control positions in trimmed forward flight were evaluated for level, climbing, and descending flight at the general conditions listed in table 2. Static trim characteristics were evaluated by trimming the aircraft in 10-knot increments from 80 to 200 KCAS and recording aircraft attitudes, control positions, and control forces. The data analysis methods used are described in appendix D. The results of this test are presented in figures 12 and 13, appendix E.

22. Rudder and aileron control positions were essentially constant for airspeeds above 100 KCAS in level flight. The elevator control required for level flight shows a shallow but positive gradient in the airspeed range from 100 to 200 KCAS, with 1.5 inches total control movement required at the light gross weight for this airspeed range and 2.0 inches required at the heavy gross weight. A comparison of the heavy weight (predrop) and light weight (postdrop) elevator control positions at 130 KCAS shows a control trim shift of 0.5 inch forward. This trim shift will be further discussed in paragraph 41. Although 12.6 inches of elevator control travel is available, only the forward 5 inches or 40 percent of the total range is used.

Static Longitudinal Stability

23. The static longitudinal stability of the TS-2A aircraft was evaluated by trimming the aircraft in coordinated level flight then varying airspeed in 5-knot increments for a total airspeed excursion of ±20 knots from trim. Control positions and control forces were recorded at each airspeed. This test was conducted under the general conditions listed in table 2 and data analysis methods used are described in appendix D. The test results are presented as figures 14 and 15, appendix E.

24. Stick-free static longitudinal stability, as indicated by the variation of elevator control force with airspeed, was essentially neutral for both the heavy and light gross weight configurations. The neutral control force gradient about trim does not provide the pilot with any force cues during airspeed changes and requires significant pilot compensation to hold a precise airspeed. Also, during a mission, as in maneuvering for a retardant load drop, the pilot's attention should be largely outside the cockpit to ensure terrain clearance, aircraft avoidance, and an accurate load drop. The lack of force cues with airspeed changes will require a diversion of attention from mission maneuvering to the cockpit (airspeed indicator) to preclude excessive airspeed buildup and subsequent severe pitch-up and high normal load factors during retardant drops, as discussed in paragraph 44.

25. Stick-fixed static longitudinal stability, as indicated by the variation of elevator control position with airspeed, is positive, but with a shallow gradient. In the light gross weight configuration, the gradient is 0.013 inch per knot. In the heavy gross weight configuration, this gradient is more positive, but still shallow, with 0.025 inch per knot. The shallow stick-fixed static longitudinal stability does not

give the pilot adequate control position cues with airspeed increase and will require a diversion of attention from outside the aircraft during mission maneuvering to preclude excessive airspeed buildup. This problem is further discussed in paragraph 39. The weak static longitudinal stability results in difficulty in maintaining precise airspeed control.

Static Lateral-Directional Stability

26. The static lateral-directional stability characteristics of the TS-2A airplane were evaluated at the conditions shown in table 2. Lack of sideslip instrumentation required that aircraft handling qualities be referred to ball position (side force). The aircraft was stabilized at a constant airspeed and at a constant heading while the ball position (side force) was varied left and right of trim. The data analysis methods are presented in appendix D. The test results are presented as figures 16 and 17, appendix E.

27. Static directional stability, as indicated by the variation of ball position with rudder pedal position, was positive. No appreciable lightening of rudder forces was noted, and the variation of rudder force and position with ball position was linear throughout the range tested. The static directional stability characteristics of the TS-2A airplane are positive and satisfactory.

28. Dihedral effect, as indicated by the variation of aileron control position displacement with ball position, was positive. Further evaluation using rudder-only turns confirmed strong dihedral effect, in that bank angle and turns could be partially controlled by small rudder displacements.

Maneuvering Stability

29. The maneuvering stability characteristics of the TS-2A aircraft were evaluated at the conditions listed in table 2. These tests were conducted by establishing the trim conditions in both left and right turns and then while maintaining airspeed and power, increasing normal acceleration incrementally to the maximum allowable. Altitude was varied as necessary to maintain airspeed. Aircraft response during symmetrical pull-up and pushover maneuvers was also evaluated. The data analysis methods used are described in appendix D. Test results are presented in figures 18 through 21, appendix E.

30. Stick-free maneuvering stability, as indicated by the variation of control force with normal acceleration, was strongly positive (increased aft elevator control force with increased load factor) and essentially linear for all conditions tested. The lowest value obtained was at light gross weight (approximately 21 lb/g) using the symmetrical pull-up technique, and was sufficiently high to prevent inadvertent control inputs which could produce undesirably abrupt aircraft response or tendencies toward pilot-induced oscillations.

31. Stick-fixed maneuvering stability, as indicated by the variation of elevator control position with normal acceleration, was positive (increased aft elevator control motion with increased load factor) and essentially linear. The lowest value of 1.5 in./g was obtained at light gross weight conditions using the symmetrical pull-up and pushover technique, and was sufficiently high to give good maneuvering control harmony. The maneuvering stability is satisfactory for the airtanker mission.

Roll Performance

32. The roll performance of the TS-2A aircraft was evaluated at the conditions listed in table 2, by performing inputs of 1/3, 2/3, and full aileron control deflection in both left and right directions. During these tests, rudder pedals were held fixed and adverse yaw was evaluated. The data analysis methods are presented in appendix D. A representative time history is presented in figure 22, appendix E. The roll rate from a full aileron control deflection was in excess of 30 degrees per second, which is satisfactory for the airtanker mission.

Single-Engine Minimum-Control Airspeed

33. Single-engine V_{MC} was evaluated at the general conditions listed in table 2. The data analysis methods are presented in appendix D. Static V_{MC} in the level flight configuration was determined with the left engine operating in a "zero-thrust" condition (minimum propeller speed and 10 in. Hg MAP) and the right engine operating at normal climb power (2300 rpm and 35 in. Hg MAP). Airspeed was slowly reduced until full control displacment was reached or stall occurred. The static V_{MC} varied from 78 KIAS with a 5-degree left bank to 74 KIAS with a 5-degree right bank. Static V_{MC} was defined by stall in each case. The stalls were relatively mild with sufficient stall warning in terms of aircraft buffet and the control shaker. Adequate aircraft recovery was easily accomplished by releasing the control yoke back pressure and reducing power on the operating engine. There was no tendency toward poststall gyrations or other adverse characteristics. The static V_{MC} characteristics of the TS-2A are satisfactory.

34. Dynamic V_{MC} was evaluated in the takeoff configuration by stabilizing the aircraft at the desired airspeed with maximum power and at NATOPS recommended takeoff trim settings. Power was rapidly reduced to IDLE on the left engine with the propeller remaining unfeathered. After approximately 1 second, recovery to straight flight was initiated. Airspeed was reduced approximately 5 knots and the test repeated until an airspeed was reached where stall occurred or straight flight could not be maintained. Using this method, dynamic V_{MC} was found to be 100 KIAS, the airspeed at which stall warning (control shaker) occurred with both engines operating at takeoff power, prior to power being reduced on the left engine. In each case, straight flight could be easily regained but relatively high rates of descent (up to 900 ft/min) occurred immediately after power was reduced on the left engine. Following an engine failure on takeoff, maximum braking should be used to stop the aircraft (see para 16) if sufficient runway remains. If sufficient airspeed has been reached, a continued takeoff could be successfully accomplished by jettisoning the load and retracting the gear, as recommended by the current

emergency procedures. At maximum gross weight conditions (26,000 pounds) the dynamic V_{MC} is 100 KIAS. This information should be included in the FAA flight manual supplement. The dynamic V_{MC} characteristics of the TS-2A are satisfactory.

Stalls

35. The stall characteristics of the TS-2A aircraft were evaluated under the general conditions and configurations listed in tables 1 and 2. Stalls were evaluated by establishing a trim condition and reducing airspeed at a rate of 1 knot per second until the stall occurred. Recovery was made following moderate aircraft buffet or actual stall departure. A discussion of the data analysis methods used is presented in appendix D. Representative time histories of light and heavy gross weight stalls are presented as figures 23 through 26, appendix E. Table 3 presents a summary of the stall data.

36. The stall characteristics presented in the NATOPS flight manual are accurate for the tanker configured TS-2A aircraft. In addition, the stall warning from the control shaker provides adequate and sufficient cues to an impending stall situation. The stall characteristics are satisfactory for the airtanker mission.

Mission Maneuvering Characteristics

37. The primary mission of the tanker configured TS-2A aircraft is to drop liquid chemical fire retardant accurately and in a controlled pattern on designated locations, generally in the path of a fire. To accomplish this task, extensive aircraft maneuvering may be required to maintain terrain clearance, and for aircraft smoke and fire avoidance. The aircraft must be properly aligned to insure an accurate load drop, and to allow for a safe exit from the drop area. The drop is generally accomplished from descending flight, sometimes in a turn, and is usually made at approximately 100 feet AGL for proper retardant concentration. The entire drop sequence requires primary pilot concentration to be outside of the cockpit to ensure safe and satisfactory performance.

38. To minimize underboost (low power, high rpm) conditions for long periods of time and to insure immediate engine response when needed, a minimum of 15 in. Hg MAP should be maintained during the drop sequence. The residual power in a descent, combined with a relatively aerodynamically clean aircraft and the poor static longitudinal stability characteristics (paras 24 and 25), can and frequently does result in excessive undetected airspeed buildup. Due to structural limitations and pitch-up tendencies during load drop (para 40), airspeed during the drop must be maintained below 130 KCAS. Qualitative evaluation during this test program indicates the desirability of a speed brake or other high-drag device for airspeed control.

Pitch-Up Tendencies

39. The TS-2A airplane is typical of most short-coupled aircraft used in the tanker mission, in that load release is usually accompanied by moderate to severe pitch-up.

Table 3. Stall Data Summary.

Configuration[1]	Power[2]	Average Gross Weight (lb)	Stall Warning Indicated Airspeed[3] (kt)	Stall Indicated Airspeed (kt)	Characteristics
Clean (LF)	On	18,500	88	83	Wing rock, buffet nose-down pitch
Clean (LF)	Off	18,500	88	83	Wing rock nose-down pitch
Clean (LF)	On	25,500	102	96	Buffet right wing drop
Clean (LF)	Off	25,500	100	94	Buffet, wing rock nose-down pitch
Clean (LF) accel (L)[4]	On	18,500	100	94	Buffet, wing rock pitch oscillation
Clean (LF) accel (R)	On	18,500	94	87	Buffet, wing rock nose-down pitch
Clean (LF) accel (L)	On	25,500	110	105	Left roll-off pitch oscillation
Clean (LF) accel (R)	On	25,500	106	104	Pitch buffet wing rock nose-down pitch
Drop (DR) full flaps	On	18,500	73	63	Buffet nose-down pitch
Drop (DR) full flaps	On	25,500	83	75	Right roll-off nose-down pitch
Drop (DR) 2/3 flaps	On	18,500	79	65	Wing rock, right roll nose-down pitch
Drop (DR) 2/3 flaps	On	25,500	89	81	Wing rock nose-down pitch
Drop (DR) full flaps	Off	18,500	73	64	Wing rock nose-down pitch
Drop (DR) full flaps	Off	25,500	83	80	Wing rock nose-down pitch
Drop (DR) 2/3 flaps	Off	18,500	75	69	Buffet, wing rock nose-down pitch
Drop (DR) 2/3 flaps	Off	25,500	89	83	Buffet, wing rock nose-down pitch

[1]Configurations are shown in table 1. Landing gear was UP for all stalls.
[2]Power ON: 2400 rpm, 20 in. Hg MAP; power OFF: 2400 rpm, idle.
[3]Stall warning: Airspeeds listed are for yoke shaker. There was aerodynamic stall warning (buffet, wing rock, etc) which occurred close to the stall airspeed.
[4]Accelerated stalls, clean configuration. (L) and (R) refer to left or right turns during stall.

Results of this testing substantiated reports from users in the field that pitch-ups in excess of 3g are commonplace during load drop maneuvers. The TS-2A load factor limits (g) are 3.0 in Navy operation and 2.5 in FAA certified operation. Both limits are routinely exceeded during normal fire tanker operations, which presents a potential for structural failure of the wings. In addition, the possibility of pilot disorientation during the uncommanded pitch-ups is high.

40. The pitch-up is the result of a number of contributing causes, some of which are listed below.

a. Trim shift: As discussed in paragraph 22, a shift of 0.5 inch in trim curves between the heavy and light gross weight condition was noted. This trim change, when evaluated in terms of the symmetrical pull-up in maneuvering stability (fig. 21, app E), is equivalent to a load factor increment of 0.25.

b. Decreased pressure along bottom of fuselage and horizontal stabilizer: As the load is dropped at airspeeds from 110 to 130 KIAS, it exits the aircraft at the same velocity and forms a relatively solid body in the airflow. The free stream air forms a flow pattern around the body of water (or retardant) creating increased velocities and reduced static pressure in the vicinity of the body of water. As the body of water decelerates due to drag and moves aft, the undersurface of the fuselage and the horizontal tail are exposed to the reduced static pressure, resulting in a nose-up pitching moment. Evidence of the low static pressure field is noted in the characteristic of the retardant to slide along the bottom of the fuselage rather than dropping clear. An associated characteristic of this phenomenon is that the decreased pressure below the area of the tail causes an increase in downwash at the tail with a corresponding decrease in angle of attack of the tail and subsequent increase in download of the tail, causing additional pitch-up. To substantiate this theory, tufting was placed outside the boundary layer on the tail of the aircraft using tuft rakes, as shown in photo 5, appendix C. High-speed chase photography and normal speed videotape were used to obtain qualitative definition of flow direction. Analysis revealed a distinct downward deflection of the tuft patterns associated with the load drop and "cloud" passage. Predictably, the magnitude of the tuft deflection decreased with increased vertical distance from the drop cloud.

41. The Navy performed an investigation of the pitch-up tendencies of the TS-2A and S-2A airplanes in 1966 (refs 10 and 11, app A), and although the Navy's investigation was focused on pitch-up during takeoff, the problems appear to be related. The Navy cited the basic cause of the strong pitch-up tendency as the "excessive static longitudinal instability" (at high power settings). As a consequence of these tests, a modification to the longitudinal control system was proposed and tested by the Navy. This modification increased the airplane nose-down control authority by a readjustment of the elevator pushrods, yielding more trailing edge-down and less trailing edge-up maximum elevator positions. The increased trailing edge-down elevator provided acceptable control authority for stabilized flight, and for pitch-up and stall conditions for the current cg range (21.5 to

27.0 percent MAC). The modification was recommended for incorporation on the TS-2A, US-2B, and S-2A1F airplanes. An investigation should be conducted to determine if the Navy's recommended control system modification has been performed on the tanker modified TS-2A airplanes. If the modification has not been performed, it should be accomplished as soon as possible. The anticipate technique (para 45) may need to be changed after the control system modification.

Load Drop Envelope

42. The load drop envelope was evaluated as a normal portion of all heavy gross weight flights, with additional flights dedicated to testing a load drop matrix. The load drop envelope was evaluated under the general conditions listed in table 2. During these tests, the primary internal load and drop medium used was water. Four retardant drops were performed to conclude the drop matrix and to evaluate the differences encountered between these two media. The retardant is heavier and more viscous than water. The data analysis methods are briefly described in appendix D. The results of this test are presented in table 4 and figure 1. Time histories of representative drops are presented in figures 27 through 33, appendix E.

43. Table 4 presents a comparison of load drop pitch-up characteristics for various airspeeds, aircraft configurations, and load drop techniques. Figure 1 shows that the pitch-up characteristics, as evidenced by the normal load factor (g), increase in severity with increasing airspeed. Additionally, the technique of dropping a full salvo (four doors at once) yields a significantly higher load factor than the currently recommended technique of two rapid one-half salvos (2x2 salvo). The use of the 2x2 salvo yields from 0.3 to 0.7 reduction in load factor when compared with the full salvo technique. The FAA supplement to the NATOPS flight manual recommended a maximum drop airspeed of 130 KIAS, which is marginal but satisfactory. The currently used 2x2 salvo technique reduces the pitch-up tendency.

44. During the load drop evaluation, a base line was determined by using a technique whereby the controls are held fixed, just prior to and during the load drop. The controls were moved only after the normal acceleration had been allowed to build and the pitch rate had started to level off (fig. 29, app E). Conversely, the anticipate technique involves placing the control yoke to the extreme forward position (a change of about 3.5 inches) immediately after the load release (fig. 30, app E). This technique counters the shift in trim control position (paras 21 and 22) and full nose-down control is used before the pitch-up has time to build. Pitch-up will still occur; however, a reduction of approximately 0.3g can be realized when compared to the control-fixed technique. The 2x2 salvo technique and the anticipate technique should be used for all drops.

45. The weak static longitudinal stability characteristics, combined with the requirement for at least 15 in. Hg MAP (paras 23 through 25 and 39), caused undetected airspeed build-up during the approach to drop. The aircraft pitch-up characteristics and associated normal acceleration or g-loading are aggravated by increased airspeed. Airspeed increases above 130 KCAS will result in exceeding

Table 4. Load Drop Matrix.

Flap Position	Normal Load Factor During Drop[1] (g) — Indicated Airspeed at Drop (kt): 110	120	130	135	140	Remarks
1/3	---	---	3.2 (F)	---	---	Full salvo: 1 drop, 4 doors
			Accel stall			
2/3	---	3.05 (F)	3.3 (F)	3.5 (F)	2.9 (A)	
			2.1 (G)			
			1.9 (G, A)			
			*3.1 (F)			
Full	2.8 (F)	2.75 (A)	---	---	---	2x2 salvo: 2 drops in rapid sequence, 2 doors
Full	2.35 (F)	2.65 (F)	3.0 (F)	2.3 (A)	---	
2/3	---	2.5 (F)	2,3 (A)	2,9 (F)	3.0 (F)	
			2.6 (F)	---		
			1.6 (G, A)	2.6 (A)		
			*2.5 (F)	---		
			*1.9 (G, A)	---		

[1]Load dropped is water, released from level flight, except where noted by an asterisk (*).
*: Retardant drop, released from wings level, slight descent (200 ft/min).
(F): Controls fixed.
(A): Anticipate.
(G): Gear down.

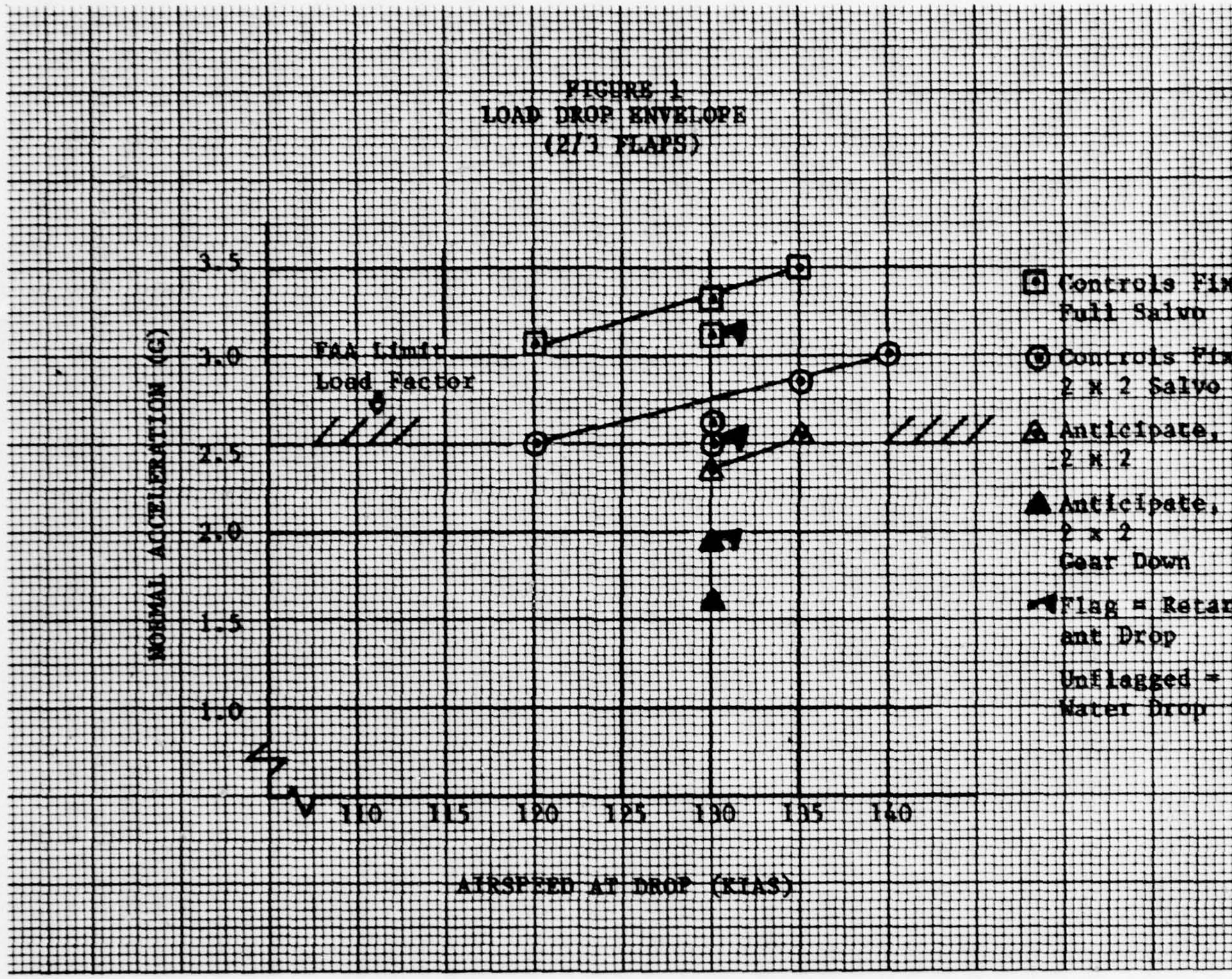


FIGURE 1
LOAD DROP ENVELOPE
(2/3 FLAPS)

the allowable (FAA limit) aircraft load factor limit (2.5) during pitch-up (depending on technique used). A speed brake or other high-drag device to help control airspeed during the drop sequence would increase the safety factor and increase the mission capability of the TS-2A (para 39).

46. Several load drops were made with the landing gear in the down position, which resulted in a number of advantages. As a high-drag device, the landing gear significantly improved airspeed control. Also, lowering the landing gear moves the aircraft cg forward (2 percent MAC) with the resulting change in the trim curve (fig. 13, app E) yielding more forward control motion available to counter the pitch-up. Additionally, at the more forward cg longitudinal static stability is slightly improved, with slightly better force and control position cues to the pilot for airspeed excursions. Table 4 shows that with gear down using the anticipate technique, 1.6g was recorded using the 2x2 salvo technique and 1.9g was recorded for the full salvo. In both cases, having the gear down and using the anticipate technique greatly reduced the pitch-up tendency, and therefore produced the lowest g loading.

47. Figures 31 through 33, appendix E, are time histories of retardant drops, and, in table 4, the drops preceded by an asterisk (*) are retardant drops (all other drops are water). The traces and the table show that having the gear down produced the lowest load factor and the least pitch-up tendency of all techniques when compared with other recommended techniques (para 44). Following the gear-down retardant drops, a thorough inspection of the landing gear area was made to determine if retardant had entered the wheel wells. No retardant was found in the main gear wells and only a small amount in the rear portion of the nosegear wheel well. Modification which would prevent the retardant from entering the nosegear wheel well to prevent corrosion and possible malfunctions of moving parts should be made. Following this modification, extended landing gear should be used during retardant drops.

48. The load factor limits for this evaluation were determined from the manufacturer's design specification and the NATOPS flight manual. The load limits for the test at 19,000 pounds gross weight were specified by the AVRADCOM airworthiness release (ref 4, app A) as 3.6 for zero, 1/3, and 2/3 flaps, and 2.5 for full flaps. This was because the flap hinge load at full flaps is much higher than at 2/3 flaps. The load factor limit for 19,000 pounds was used for the drop test since aircraft gross weight was below 19,000 pounds before the pitch-up occurred. The use of 2/3 flaps for the drop sequence is the best compromise between flap hinge load and mission requirements. Consideration should be given to increasing the limit load factor for retardant drop operations.

CONCLUSIONS

49. *Within the scope of this test, the tanker equipped* **TS-2A aircraft is marginal, but satisfactory for the airtanker mission. The pitch-up tendency during load drop is a potentially dangerous problem; however, techniques were devised that can reduce and/or compensate for the tendency. The information contained in the NATOPS flight manual as modified by the FAA supplement is, with few exceptions, appropriate for the TS-2A aircraft in the airtanker mission.**

RECOMMENDATIONS

50. A rotation airspeed of 100 KIAS and a lift-off airspeed of 105 KIAS should be used and changes to the FAA flight supplement should be made for heavy gross weight operations (para 12).

51. The accelerate/stop performance information contained in figures 6 through 11, appendix E, should be included in the flight supplement along with the following NOTE (para 17).

NOTE

> The decision of whether to retain or jettison the load in the event of an aborted takeoff should be based on judgment and precise knowledge of a number of factors, including runway length, runway condition, overrun, density altitude, and aircraft condition and configuration. The decision on abort procedures should be made prior to starting the takeoff. Although jettisoning the load may result in less runway distance used for braking (up to about 400 feet at lower density altitudes), *jettisoning the load also increases the possibility of fouling the* gear with brake lock-up and hydroplaning.

52. At maximum gross weight conditions (26,000 pounds) dynamic V_{MC} is 100 KIAS. This information should be included in the FAA flight supplement (para 34).

53. An investigation should be accomplished to determine if the Navy's recommended control system modification has been performed on the tanker modified TS-2A airplanes. If the modification has not been performed, it should be accomplished as soon as possible (para 41).

54. The 2x2 salvo technique and the anticipate technique should be used for all drops (para 44).

55. Following modifications made to preclude retardant contamination of the nose wheel well, the extended landing gear should be used during retardant drops (para 47).

56. Consideration should be given to increasing the limit load factor for retardant drop operations (para 48).

APPENDIX A. REFERENCES

1. Final Report, US Army Aviation Systems Test Activity, Project No. 71-16, *Phase 1, TS-2A Airtanker Evaluation*, July 1972.

2. Letter, AVSCOM, DRSAV-EQI, 15 Feb 1977, subject: Evaluation of TS-2A Forest Service Aircraft.

3. Technical Manual, NATOPS Flight Manual, *Navy Model TS-2A, US-2A/B/C*, December 1972.

4. Letter, AVSCOM, DRSAV-EQA(A), 4 May 1977, subject; TS-2A Airtanker Evaluation Phase II Safety-of-Flight Release (SOFR).

5. Flight Test Manual, Naval Air Test Center, FTM No. 104, *Fixed Wing Performance*, 28 July 1972.

6. Flight Test Manual, Naval Air Test Center, FTM No. 103, *Fixed Wing Stability and Control*, 4 August 1969.

7. Handbook, Air Force Test Pilot School, FTC-TIH-70-1001, *Performance*, September 1970.

8. Handbook, Air Force Test Pilot School, FTC-TIH-68-1002, *Stability and Control*, September 1968.

9. Supplement to NATOPS Flight Manual FAP, *Navy Model TS-2A, US-2A/B/C*, 2 August 1974.

10. Technical Report, Naval Air Test Center, No. FT-334-66, *Investigation of Pitch-up Tendencies in the TS-2A and S-2A Airplanes*, April 1966.

11. Technical Report, Naval Air Test Center, No. FT-76R-66, *Investigation of Pitch-up Tendencies in the TS-2A and S-2A Airplanes*, August 1966.

APPENDIX B. AIRCRAFT DESCRIPTION

1. The Grumman TS-2A was originally designed for the Navy for use as a primary instrument and dual-engine trainer. It and other versions of the S-2 series were used for the antisubmarine warfare mission in the late 1940's. Surplus TS-2A aircraft have been released by the Navy to the Forest Service for modification and use in the airtanker mission.

2. Conversion of the TS-2A to an airtanker reduced the crew requirement from two to one and required the installation of a tank system described beginning with paragraph 20. To lighten the aircraft as much as possible, all mission nonessential equipment was removed to include wing-fold system, arresting hook system, g-limiter, auto pilot, torpedo bay doors and systems, pneumatic deicer boots, antifog and windshield defrost systems, searchlight, over-water emergency equipment, and the cabin heat system. Photos 1 and 2 show two external views of the aircraft and photos 3 and 4 show the current cockpit arrangement and overhead control panel. The physical dimensions of the aircraft are listed below.

Dimensions

Wing span	69 ft, 8 in.
Horizontal stabilizer span	22 ft, 10 in.
Length	42 ft
Height to top of vertical stabilizer	16 ft, 3.5 in.
Propeller diameter	11 ft
Propeller/fuselage clearance	5 in.
Propeller/ground clearance	1 ft, 10-3/4 in.
Distance between main gear	18 ft, 4 in.
Distance between main and nose gear	13 ft, 5 in.

Wing Area

Wing area	466.4 ft^2
Aileron area	18.6 ft^2
Flap area	97.7 ft^2

Weights

Maximum takeoff weight	27,000 lb
Maximum landing weight	24,500 lb
Maximum zero fuel weight	24,000 lb

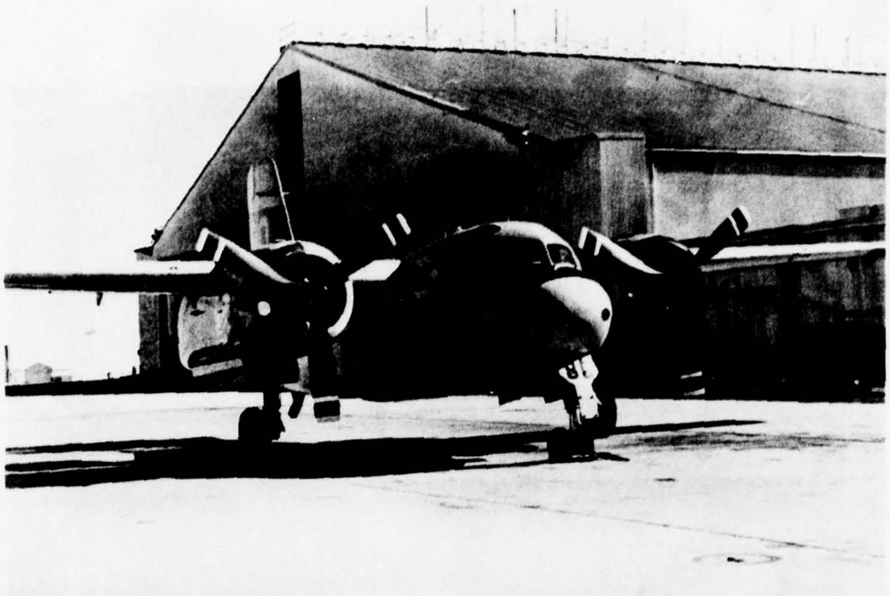


Photo 1. Three-quarter Right Side View.

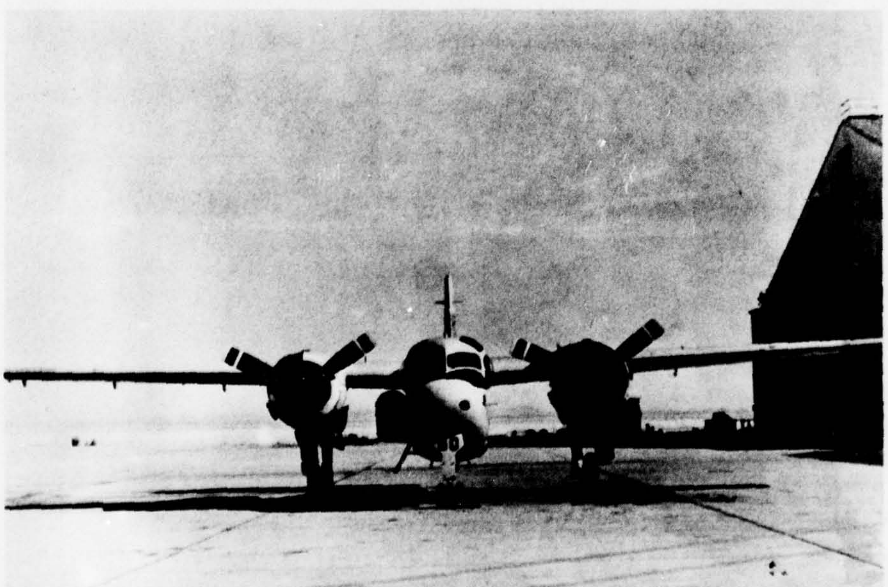


Photo 2. Front View.

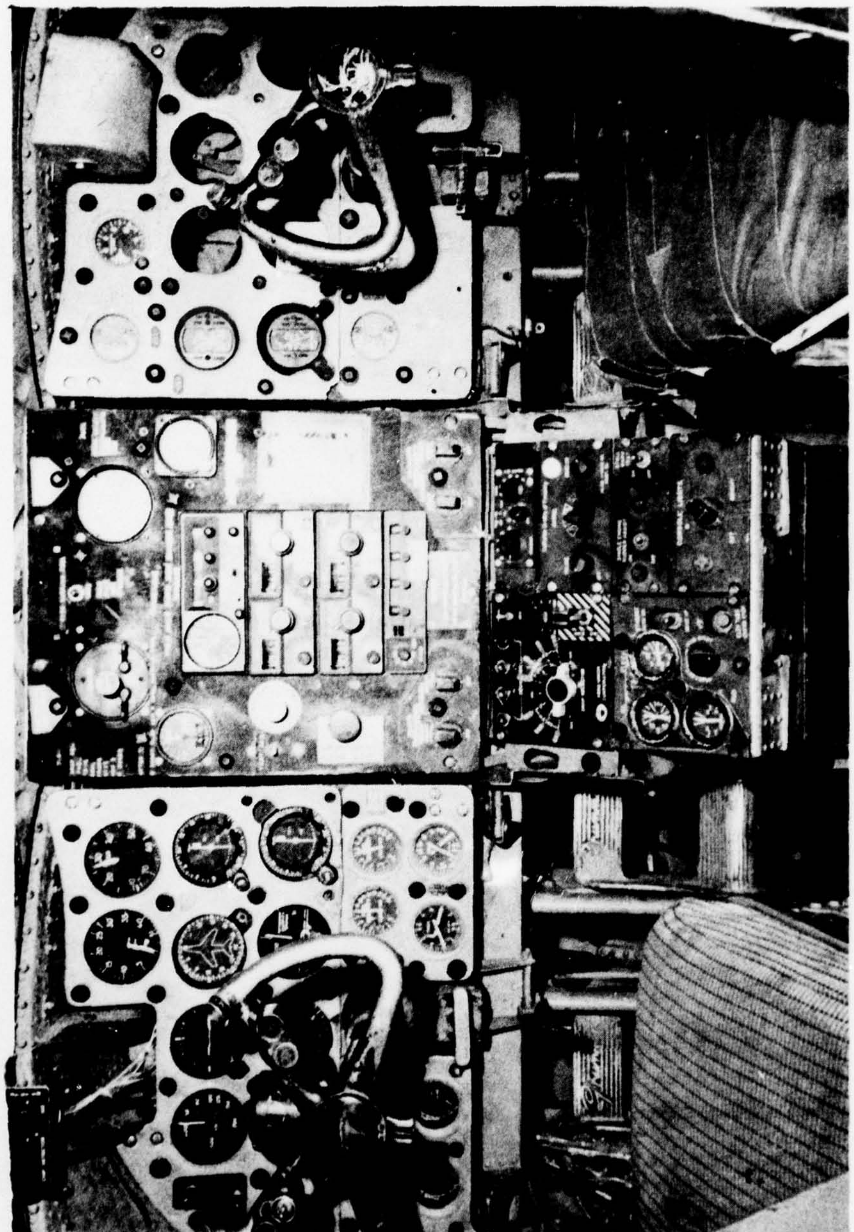


PHOTO 3. PILOTS STATION

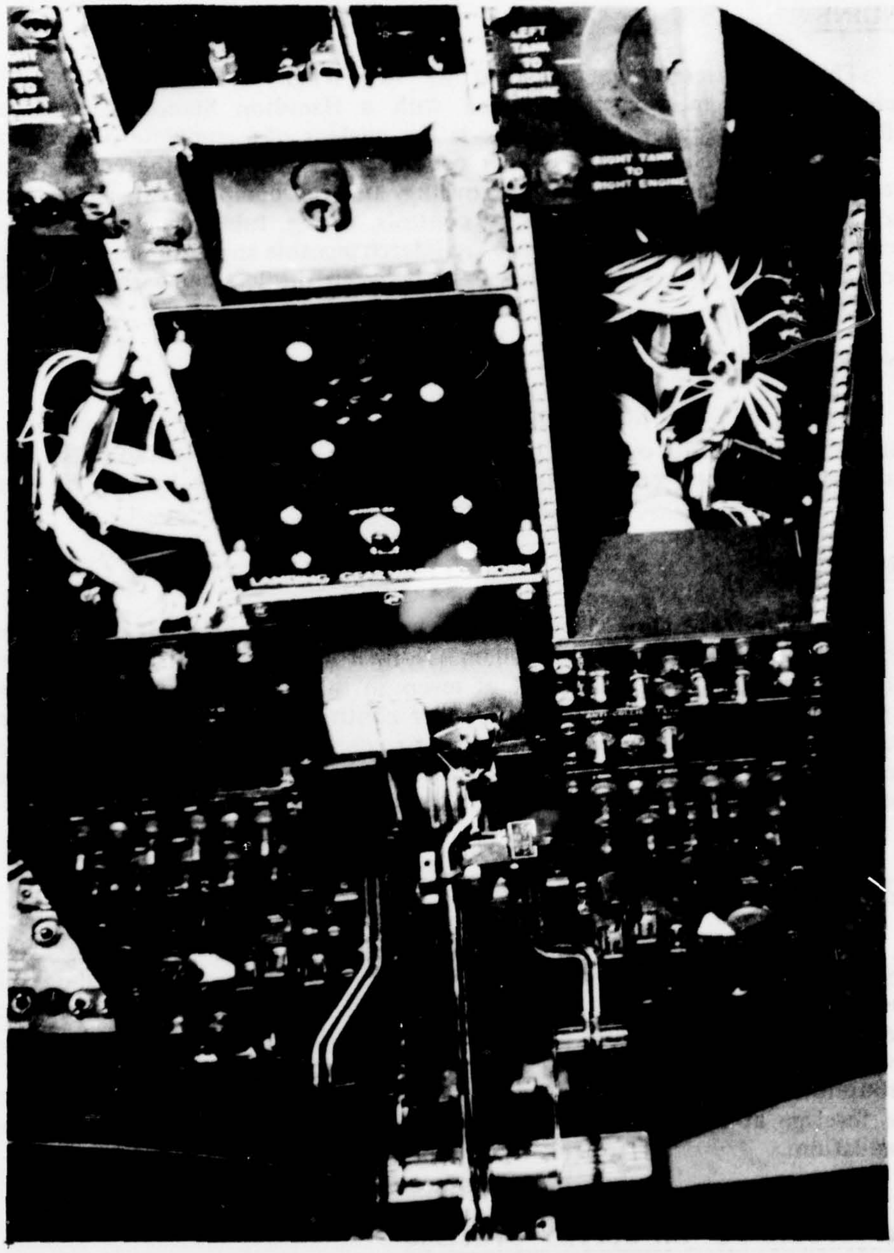


Photo 4. Overhead Control Panel.

ENGINE

3. The test aircraft was powered by two Wright R1820-82A nine-cylinder reciprocating engines, each equipped with a Hamilton Standard three-bladed, full-feathering propeller. Each engine is the nucleus of a complete power package or quick engine change unit, which consists of a propeller, the engine with all accessories, the engine mount and cowling, and the complete oil, air, induction, and exhaust systems. The necessary controls, wiring, tubing, and connectors are included. The complete engine units are interchangeable and only minor differences such as routing of hoses and lines exist between the left and right installations.

ENGINE CONTROLS

4. Dual-control levers for the throttle, mixture, and propeller are contained in a quadrant in the center overhead panel (photo 4). Flexible cables lead to the firewall and from there bell cranks, pushrods, and flexible controls lead to the engine.

PROPELLERS

5. The aircraft is equipped with two 11-foot diameter, three-blade, full-feathering, constant-speed hydromatic propellers. Normal operation of the propellers is controlled manually by movement of levers in the cockpit. A propeller governor is mechanically connected to the propeller control handles on the engine control quadrant (photo 4). The propeller governor supplies engine oil under pressure to the propeller mechanism for propeller pitch control. Feathering and unfeathering operations are accomplished through a button-controlled feathering pump.

FUEL SUPPLY SYSTEM

6. Fuel is carried in two tank systems consisting of three cells each. The cells are located in the wing center section and in the wing on each side of the fuselage. Fuel can be fed from either tank through the tank fuel line manifold, then through the tank selector valve, the emergency fuel shutoff valve, the fuel strainer, the electrically-driven auxiliary fuel pump and the engine-driven fuel pump to the carburetor. Cross-feed is provided by a line running from each manifold across the fuselage at the rear beam to the tank selector valve on the opposite engine installation.

HYDRAULIC POWER SUPPLY SYSTEMS

7. The normal hydraulic system is of the engine-driven, variable volume-pump, closed-center type. It provides an operating pressure of 1500 psi for landing gear, wing flaps, rudder assist, retardant tank doors, and brakes. Pressure to the 1500-psi systems is supplied directly during normal flight. A main reservoir (3.7 gallons)

and an emergency reservoir (1.3 gallons) are provided, with the latter being refillable in flight. The emergency system consists of a reservoir, hand pump, and selector valves to operate the landing gear and flaps down only.

OIL SUPPLY SYSTEM

8. The oil supply system is a self-contained unit within the power package. Each engine oil supply system consists of a supply tank, an oil cooler and all necessary tubing, fittings and valves to provide engine lubrication, oil dilution, tank and engine venting, draining, and propeller feathering. Oil for propeller feathering is taken from the bottom of the oil tank and run through a hose to the feathering pump. The oil tank has a capacity of 17 gallons (12.5 usable gallons) and is mounted forward of the firewall between the upper engine mount tubes.

LANDING GEAR SYSTEM

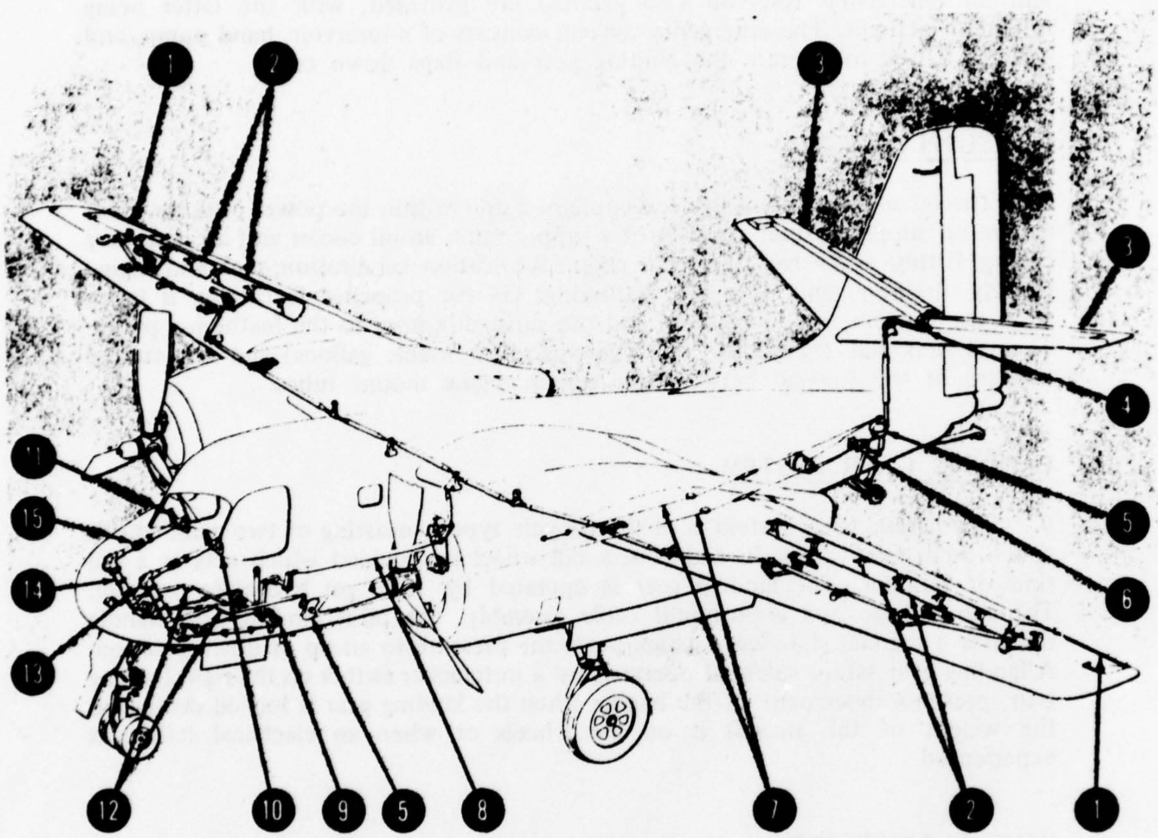
9. The landing gear system is of the tricycle type, consisting of two main wheels and a dual nose wheel. In addition, a tail wheel is provided which acts as a tail skid or bumper. The landing gear is operated by 1500 psi hydraulic pressure. Through linkage and a push-pull cable assembly, the pilot landing gear handle operates a manual slide valve which ports the pressure to an up or down position. A landing gear safety solenoid operated by a nutcracker switch on the right landing gear, prevents movement of the handle when the landing gear is locked down and the weight of the aircraft is on the wheels or when an electrical failure is experienced.

WING FLAP SYSTEM

10. Conventional wing flaps, hydraulically actuated by 1500 lb/in.^{2} (psi) pressure, are hinged to the trailing edge of the wing. The pilot's wing flap handle positions a four-way selector valve, by means of cables, to set the flaps at full up, 1/3 down, 2/3 down and full down. The flap system is equipped with a position indicator.

FLIGHT CONTROL SYSTEM

11. The primary flight control surfaces are operated from either the pilot or copilot position by means of a control wheel and two rudder pedals which are mechanically connected to the control surfaces. All control surfaces are conventional except that a rudder trimmer (para 15) is installed for directional trimming. A rudder assist (para 16) is provided for additional directional control in the event of failure or excessive loss of power of either engine. Spoilers in each wing are mechanically linked with the ailerons as part of the lateral control system. Aileron, elevator,



1. AILERON
2. SPOILERS
3. ELEVATOR
4. BELLCRANK ASSEMBLY
5. ELEVATOR CONTROL SECTOR
6. DOWN-SPRING MECHANISM
7. CONTROL CABLE
8. PUSHROD ASSEMBLY
9. SECTOR TUBE ASSEMBLY
10. PILOT'S CONTROL WHEEL
11. CO-PILOT'S CONTROL WHEEL
12. BELLCRANK
13. SPROCKET
14. CONTROL COLUMN
15. CO-PILOT'S CONTROL DISCONNECT

Figure 1. Lateral and Longitudinal Control System.

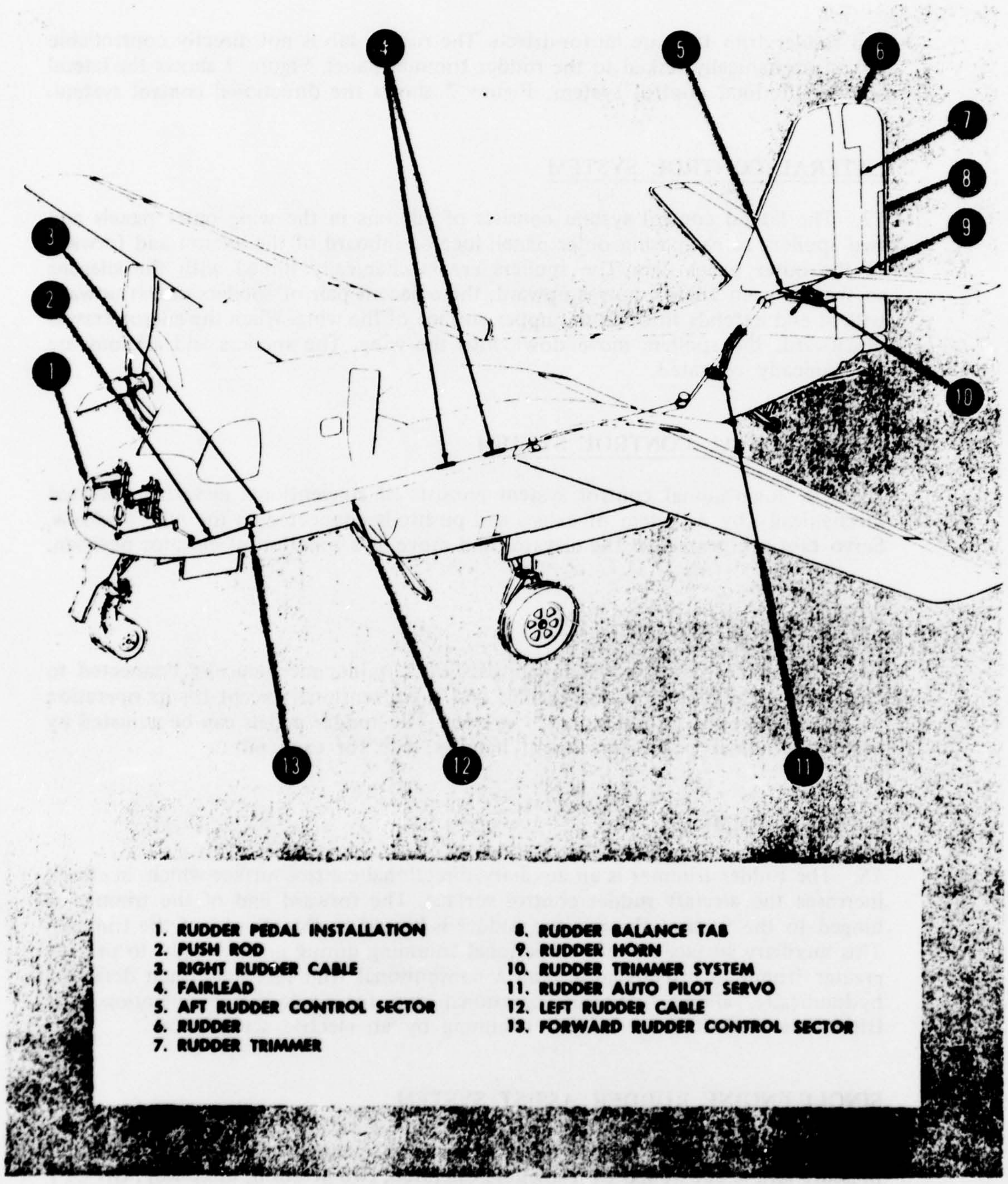


Figure 2. Rudder Control System.

and rudder trim tabs are motor-driven. The rudder tab is not directly controllable but is mechanically linked to the rudder trimmer panel. Figure 1 shows the lateral and longitudinal control system. Figure 2 shows the directional control system.

LATERAL CONTROL SYSTEM

12. The lateral control system consists of ailerons in the wing outer panels and two spoilers in each wing outer panel, located inboard of the aileron and forward of the outer panel flap. The spoilers are mechanically linked with the ailerons so that as each aileron moves upward, the adjacent pair of spoilers moves upward with it and extends through the upper surface of the wing. When the aileron travels downward, the spoilers move down into the wing. The spoilers and ailerons are mechanically operated.

LONGITUDINAL CONTROL SYSTEM

13. The longitudinal control system consists of conventional elevators operated mechanically by a system of cables and pushrods connected to the pilot's wheels. Servo tabs are geared to the elevator and move as a function of elevator position.

RUDDER CONTROL SYSTEM

14. The rudder control system consists of a rudder mechanically connected to two pairs of adjustable rudder pedals and is conventional except for its operation with the single-engine rudder assist system. The rudder pedals can be adjusted by using the rudder pedal adjustment handles, one for each pilot.

RUDDER TRIMMER

15. The rudder trimmer is an auxiliary directional control surface which, in effect, increases the aircraft rudder control surface. The forward end of the trimmer is hinged to the vertical fin and the rudder is hinged to the aft end of the trimmer. This auxiliary surface permits directional trimming during normal flight to provide greater trimming effectiveness than a conventional trim tab, and when deflected hydraulically, allows sufficient directional trim for single-engine operation. The trimmer is deflected for normal trimming by an electric screw jack.

SINGLE-ENGINE RUDDER ASSIST SYSTEM

16. The single-engine rudder assist system is operated by 1500 psi hydraulic pressure and is controlled by a solenoid-operated, two-position, three-way, ON-OFF selector valve. Two concentric slide valves are connected to the rudder control system through a bell crank and followup rod. A rudder trimmer control cylinder,

actuated by hydraulic fluid through the concentric slide valves, moves the rudder trimmer left or right, corresponding to rudder pedal movement. A rudder trimmer lock cylinder locks the system for normal flight. The rudder trimmer hydraulic system selected by the SINGLE ENGINE RUDDER ASSIST switch can be manually turned ON or OFF. With RUDDER ASSIST switch OFF, the rudder trimmer can be actuated 5-1/4 degrees left or right by an electric screw jack. For emergency use (SINGLE ENGINE RUDDER ASSIST switch ON), the rudder trimmer is deflected by a hydraulically operated servomechanism. This system increases directional control with conventional use of the rudder pedals. A follow-up system positions the trimmer proportionally to rudder deflection.

TRIM CONTROLS

17. An electrically actuated trim tab is provided on the left aileron and on each of the elevators. Rudder trimming is accomplished through an electric screw jack; control of the rudder trimmer is accomplished by the rudder trim switch located on the lower center console. A tab located on the rudder is not directly controllable, but serves only as a slave to the rudder trimmer and operates with it.

YOKE TRIM TAB CONTROL BUTTONS

18. A momentary contact switch for control of the aileron and elevator trim tabs is located on both the pilot and copilot control wheels. With the switch in the forward or aft position, the elevator trim tab is changed until the switch is released. The same switch is used to control aileron trim by moving it left and right.

TRIM TAB POSITION INDICATORS

19. Three trim tab position indicators are on the trim tab control panel on the diagonal console. They show the setting of elevator tabs, aileron tab, and rudder trimmer in degrees of movement.

STALL WARNING SYSTEM

20. An electric stall warning system gives warning of an impending stall by a control shaker mounted on the pilot yoke. The system gives an accurate prestall warning regardless of flap settings, power settings, accelerations, and configurations by automatically compensating for these conditions. The system consists of a lift computer, a yoke shaker, an adjustment box (which is part of the computer), and a test switch. As the aircraft approaches stall airspeed, the small vane of the lift transducer, located on the leading edge of the left wing, senses the angle of attack and closes the circuit through the adjustment box to the control shaker. The lift computer compensates for thrust and flap setting, which is detected by the flap position potentiometer, and relays the current to the control shaker motor.

The motor drives an eccentric mass which, in turn, vibrates the yoke as a warning signal. Under different flight configurations, the yoke shaker will operate at the following different airspeeds:

Power approach (19,000 lb)	70 KIAS ±3
Landing (19,000 lb)	75 KIAS ±3
Cruise (22,500 lb)	91 KIAS ±4

RETARDANT TANK AND DROP SYSTEM

General Description

21. Aero Union Corporation's Grumman S-2 tank system can hold up to 800 gallons of fire retardant mix. Contained in four separately controlled drop compartments, the retardant can be dispersed in several different patterns, as illustrated in figure 3.

22. The tank compartments are filled from the tail of the airplane through a single quick-disconnect fitting located just below the rudder. Quantity received is indicated to both ground crew and pilot, each being furnished a gauge, one in the cockpit and the other near the filler in the tail.

23. The tank doors are opened and closed by hydraulic actuators controlled electronically through a relay system. The airplane's 1500-psi hydraulic system supplies the energy. The system has provisions to automatically or manually select different drop configurations. One, two, or all four compartments at a time can be selected, as shown in figure 4.

Drop System Control Panel

24. The drop system control panel (fig. 5) is located on the lower portion of the center instrument panel. It contains the following:

- Drop system arming switch
- Four door position lights
- Drop selector dial and switch
- Auto-manual switch
- Short-long switch

Drop System Control Panel

25. This switch has two positions, ARM and OFF, and is guarded in the armed position. For normal operation of the doors, this switch must be in the armed position.

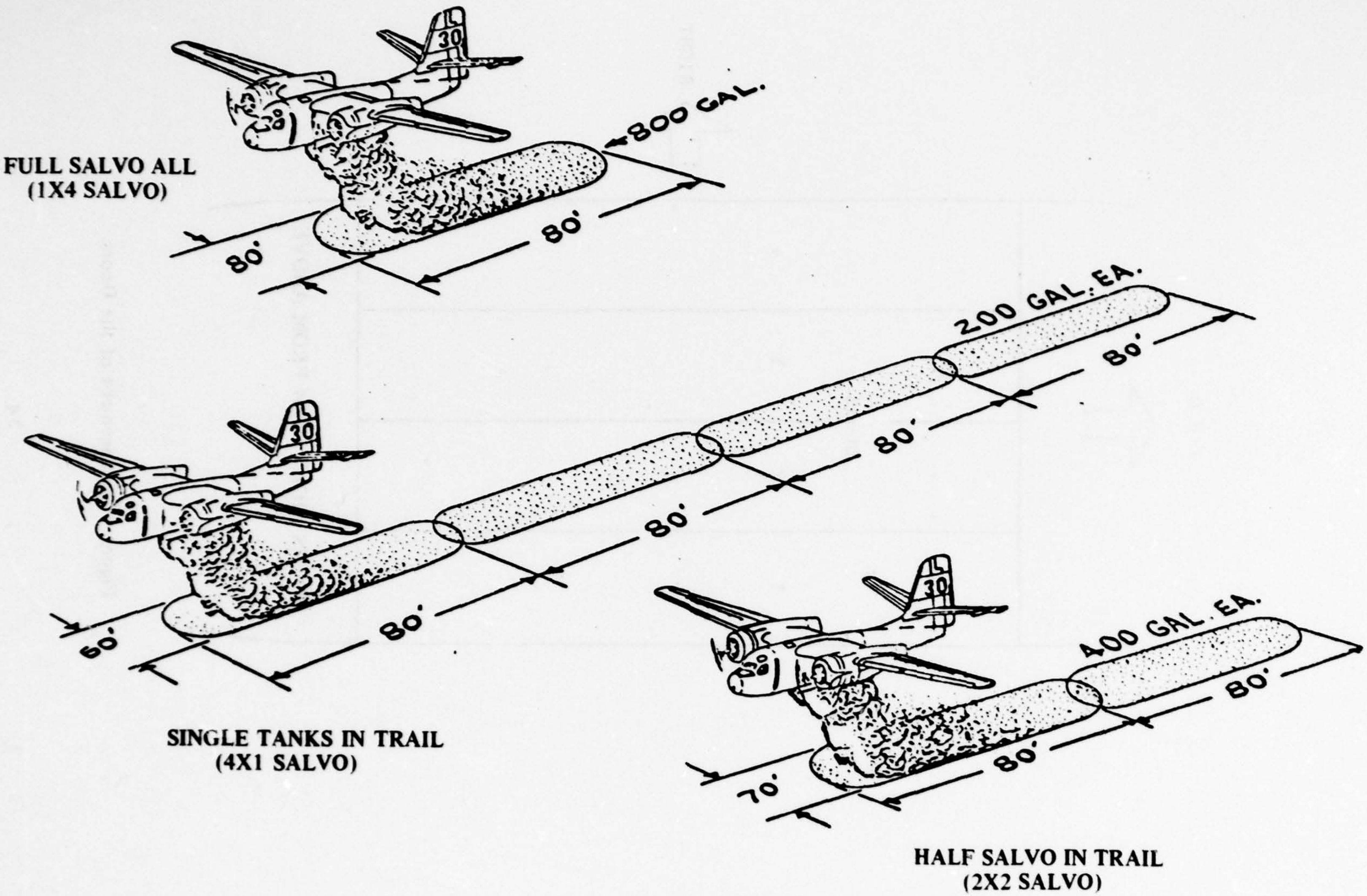


Figure 3. Possible Drop Patterns.

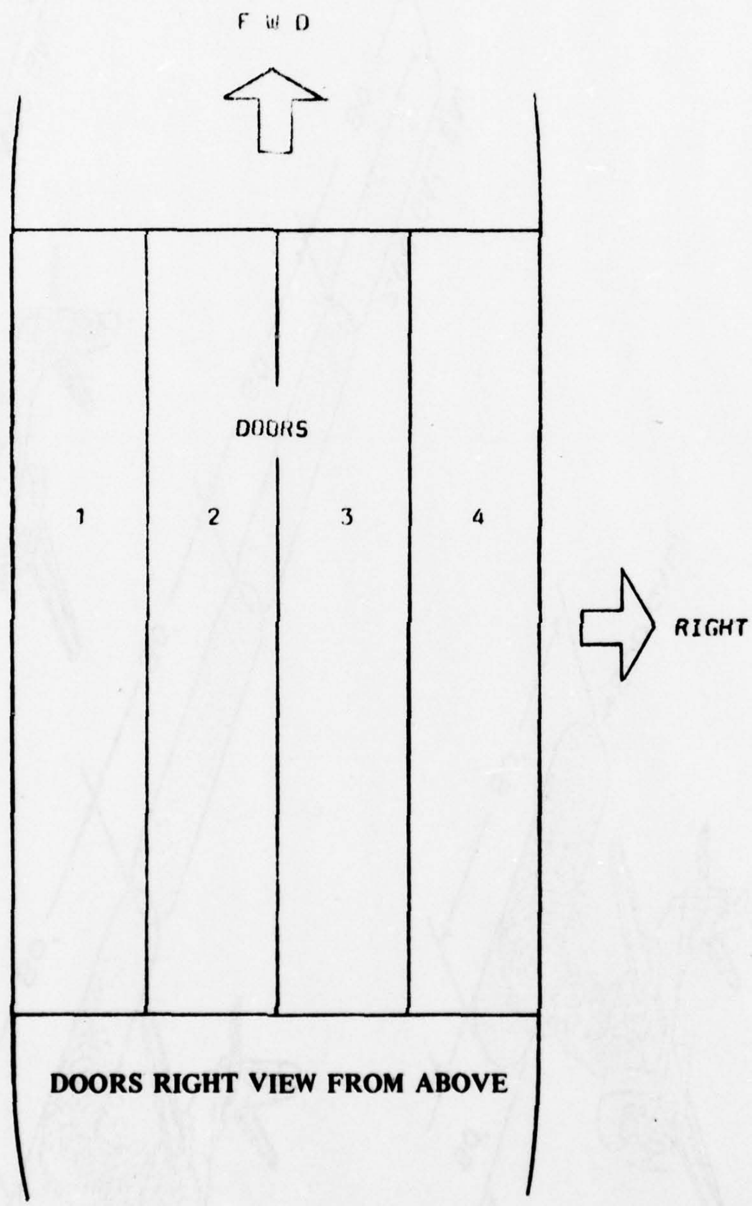


Figure 4. Arrangement of the Doors.

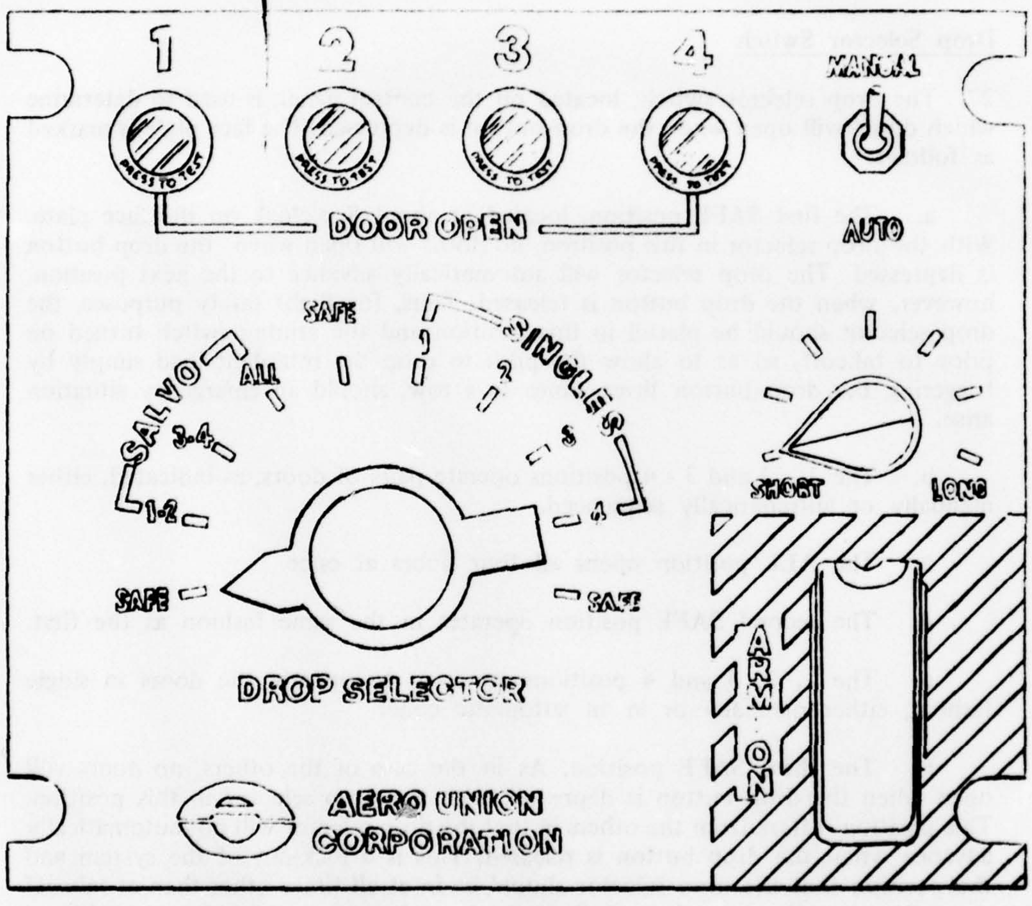


Figure 5. The Drop System Control Panel.

Door Position Lights

26. There is one amber-colored light for each of the four tank doors. Each light comes on when its respective door is in the open position, staying lit until the closing door actuates a limit switch in the fully closed and locked position.

Drop Selector Switch

27. The drop selector switch, located on the control panel, is used to determine which doors will open when the drop button is depressed. The face plate is marked as follows:

a. The first SAFE position, located at about 8 o'clock on the face plate. With the drop selector in this position, no doors will open when the drop button is depressed. The drop selector will automatically advance to the next position, however, when the drop button is released. Thus, for flight safety purposes, the drop selector should be placed in this position and the arming switch turned on prior to takeoff, so as to allow the pilot to drop the retardant load simply by triggering the drop button three times in a row, should an emergency situation arise.

b. The 1 - 2 and 3 - 4 positions operate pairs of doors, as indicated, either manually or automatically sequenced.

c. The ALL position opens all four doors at once.

d. The second SAFE position operates in the same fashion as the first.

e. The 1, 2, 3 and 4 positions are used to operate the doors in single fashion, either manually or in an automatic code.

f. The third SAFE position. As in the case of the others, no doors will open when the drop button is depressed with the drop selector in this position. This position differs from the others in that the drop selector will not automatically advance when the drop button is released. This is a lock-out of the system and the position that the drop selector should be in at all times other than at takeoff or while preparing to make retardant drops, to prevent inadvertent opening of the doors.

28. In summary, each time the drop button is depressed, the selected doors will open and the drop selector will automatically rotate clockwise to the next position. When the drop button is released, the doors will close after the time delay interval, built into the operating cycle, has passed.

Auto-Manual Switch

29. With this switch in MANUAL position, the drop button must be depressed and released for each new door selection. When set on AUTO the drop button need only be depressed once and held to go through a preselected sequence automatically.

Short-Long Switch

30. When the Auto-Manual switch is set on AUTO, the setting of the short-long switch will determine the time interval between separate drop events. If the switch is set in the SHORT position the time interval between door openings will be less than it would be if the switch is set on the LONG position.

Go Light

31. The green GO light is located in the pilot glare shield area along with the engine chip detector lights. This indicator shows drop system readiness. To be on, the arming switch must be in the armed position, the drop selector must be in a position other than safe and the system hydraulic pressure must be at least 1000 psi.

Drop Button

32. The drop button is a guarded pushbutton switch located on the left hand grip of the pilot control wheel. This switch triggers the door operating mechanisms as programmed on the drop selector.

Electrical System Protection and Control Devices

33. The electrical portions of the drop system are protected by two circuit breakers mounted on the circuit breaker panel behind the copilot seat. The junction box and relay panel is located on the forward side of the aft cabin bulkhead, at the rear of the retardant tank.

Hydraulic System

34. Under normal conditions, the drop system is hydraulically actuated, utilizing the ship's 1500-psi hydraulic system to supply fluid flow and return. The operating pressure is shown on gauges located on the overhead panel in the cockpit.

35. To supply enough volume for uniform operation of doors in all sequences a large accumulator has been incorporated. Ship's system pressure fills the accumulator which has been precharged with nitrogen and is directed to four electrically controlled 4-way selector valves by means of a pressure manifold. Each compartment of the tank has its own 4-way selector valve and actuating cylinder so that when a selection is made and triggered, ready pressure is routed through the valve(s) to the respective actuating cylinder(s), opening and then closing the door(s). A 2-way restricter is located in the upper lines controlling opening and closing speed. Return fluid is accepted back through the valve(s) and channeled to the ship's reservoir by way of the return manifold.

36. The lower working lines, one for each actuator, contain a shuttle valve separating the normal hydraulic system from the emergency system and is located close to the actuator. To prevent damage to the doors, the 2-way restricter also controls the door opening speed when the emergency system is used.

Hydraulic System Nitrogen Precharge

37. The large accumulator used in the normal drop system is divided by a floating piston. The lower chamber is charged with 1000 psi of nitrogen to provide a large volume of hydraulic fluid under pressure to the upper chamber for use in the drop system. This additional volume may be needed if the ship's hydraulic pumps cannot provide sufficient fluid during peak demands. The cushion of nitrogen also acts to dampen surges in the system. The filler valve and pressure gauge are located on the left-hand side of the empennage.

Emergency System

38. Should the normal system fail, an emergency pneumatic system is provided. This system consists of a forward-mounted nitrogen supply bottle charged to 1000 psi. A pressure gauge and emergency dump button are located on the instrument panel. Four shuttle valves, one for each actuator, are located in the hydraulic system between the 4-way valve and lower port of the actuating cylinders.

39. When the emergency drop button is depressed, nitrogen passes from the supply bottle through the drop valve and continues through a manifold to the shuttle valves and into the lower ports of the actuating cylinders, opening all four doors at once. The hydraulic fluid on the opposite side of the pistons within the actuating cylinders passes back through a 2-way restricter and each 4-way valve to the ship's reservoir. Speed of door opening on emergency mode is controlled by the 2-way restricter. The shuttle valves are automatic resetting and require no adjustment after use of the emergency system.

APPENDIX C. INSTRUMENTATION

1. Instrumentation was calibrated, installed, and maintained in the test aircraft by USAAEFA personnel. A magnetic tape system was used as the primary means of obtaining engineering flight data. The instrumentation package was located in the main cabin area at FS 169.5. Photo 1 shows the instrument package prior to installation and photo 2 shows the instrumentation package in place. The pilot station was left in the standard configuration with sensitive airspeed, strain-gauged control yoke (photo 3), and strain-gauged pedals (photo 4) installed. At the copilot station (photo 5) flap position, sensitive normal accelerometer, and sensitive free air temperature gauges were installed. Voice tape and a stopwatch were used as required.

2. During the takeoff and landing performance and the accelerate/stop performance tests, a recording observation instrument (ROI) was used (photo 6). This system recorded on tape the azimuth and altitude bearings to the test aircraft. The calibrated instrumentation and related special equipment installed for the evaluation are listed below:

<u>Cockpit Station</u>

Center-of-gravity normal acceleration
Airspeed
Altitude
Free air temperature
Flap position
Time of day
Run number
Rate of climb
Engine manifold pressure (both engines)
Engine propeller speed (both engines)

<u>Digital (PCM) Data Parameters</u>

Control position:
- Longitudinal
- Lateral
- Directional

Control force:
- Longitudinal
- Lateral
- Directional

Attitude:
- Pitch
- Roll
- Yaw

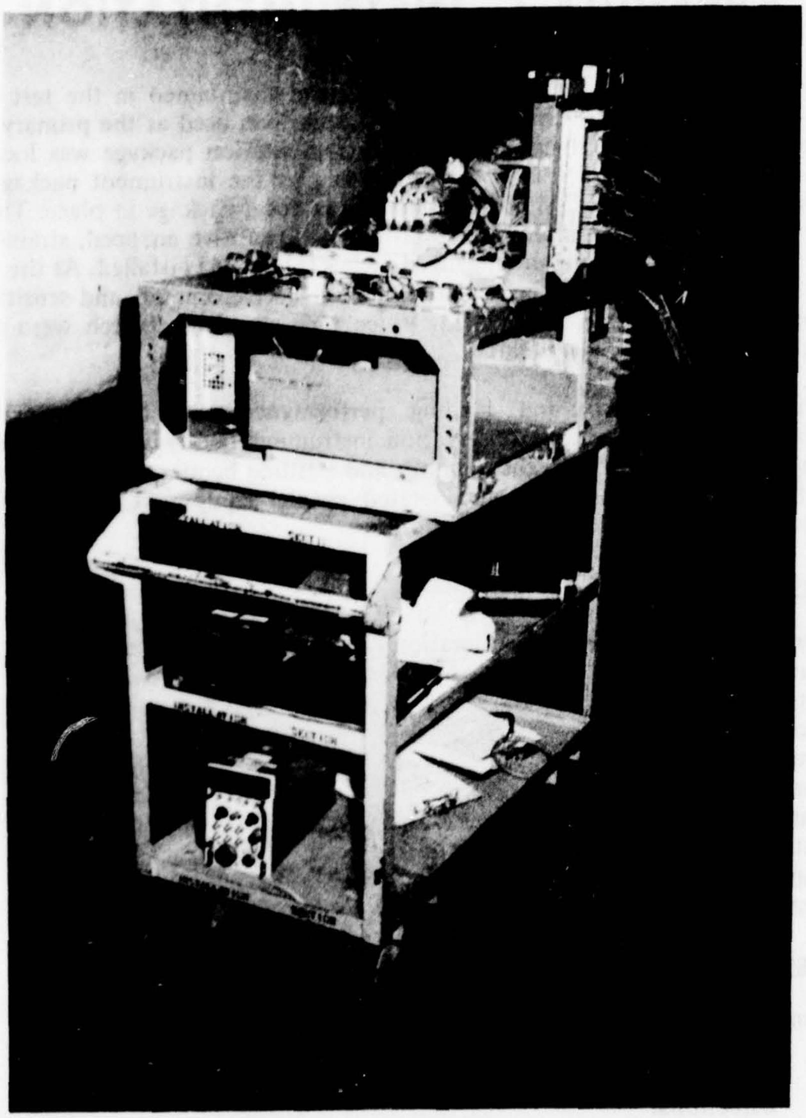


Photo 1. Instrumentation Package.

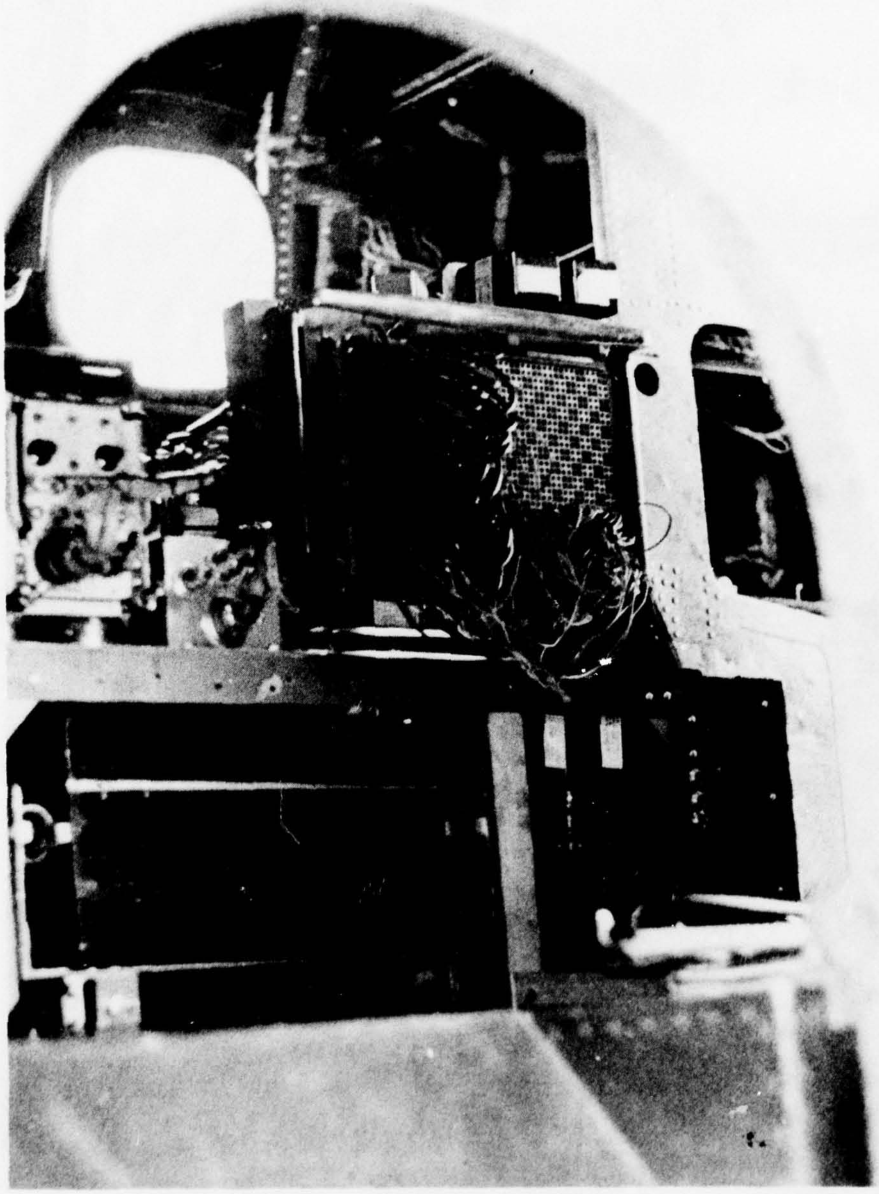


Photo 2. Installed Instrumentation Package.

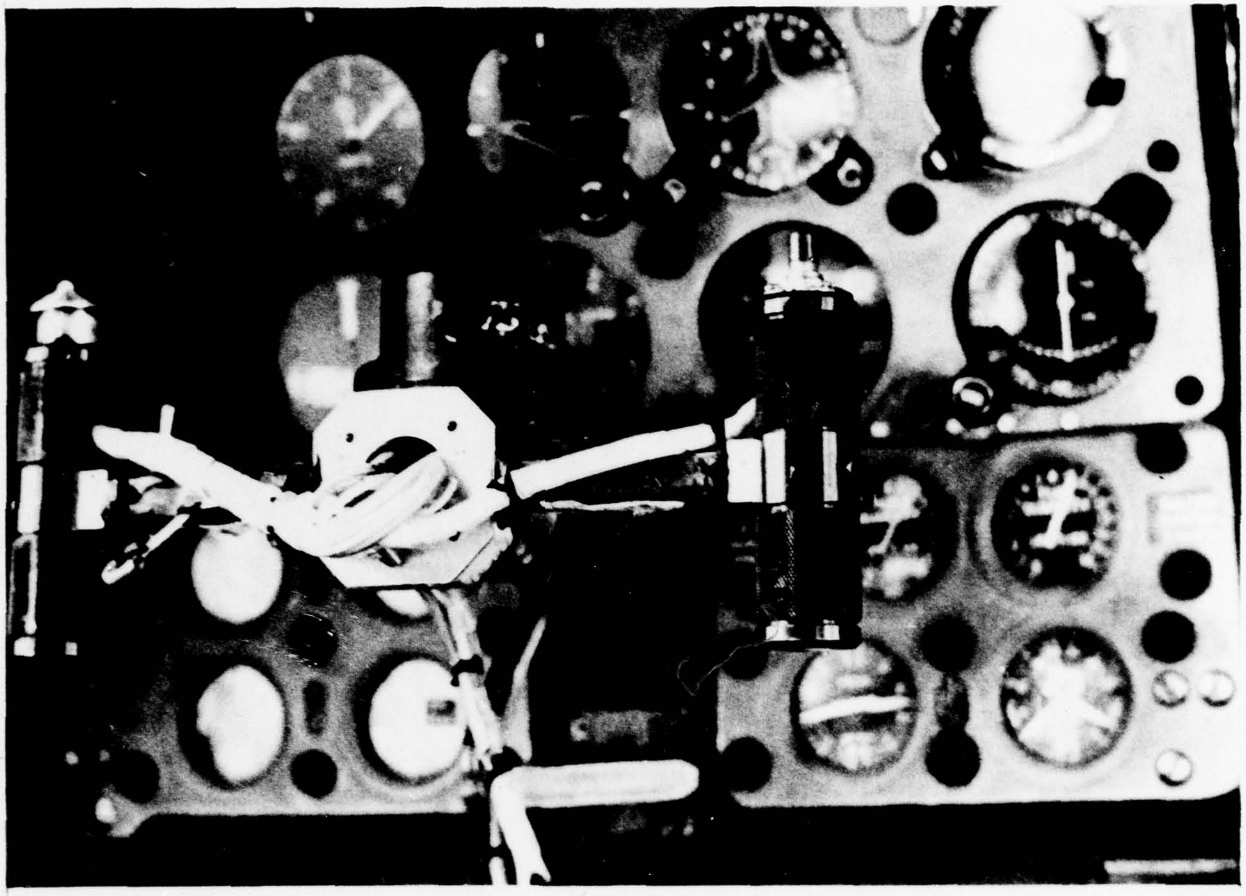


Photo 3. Strain Gaged Control Yoke.

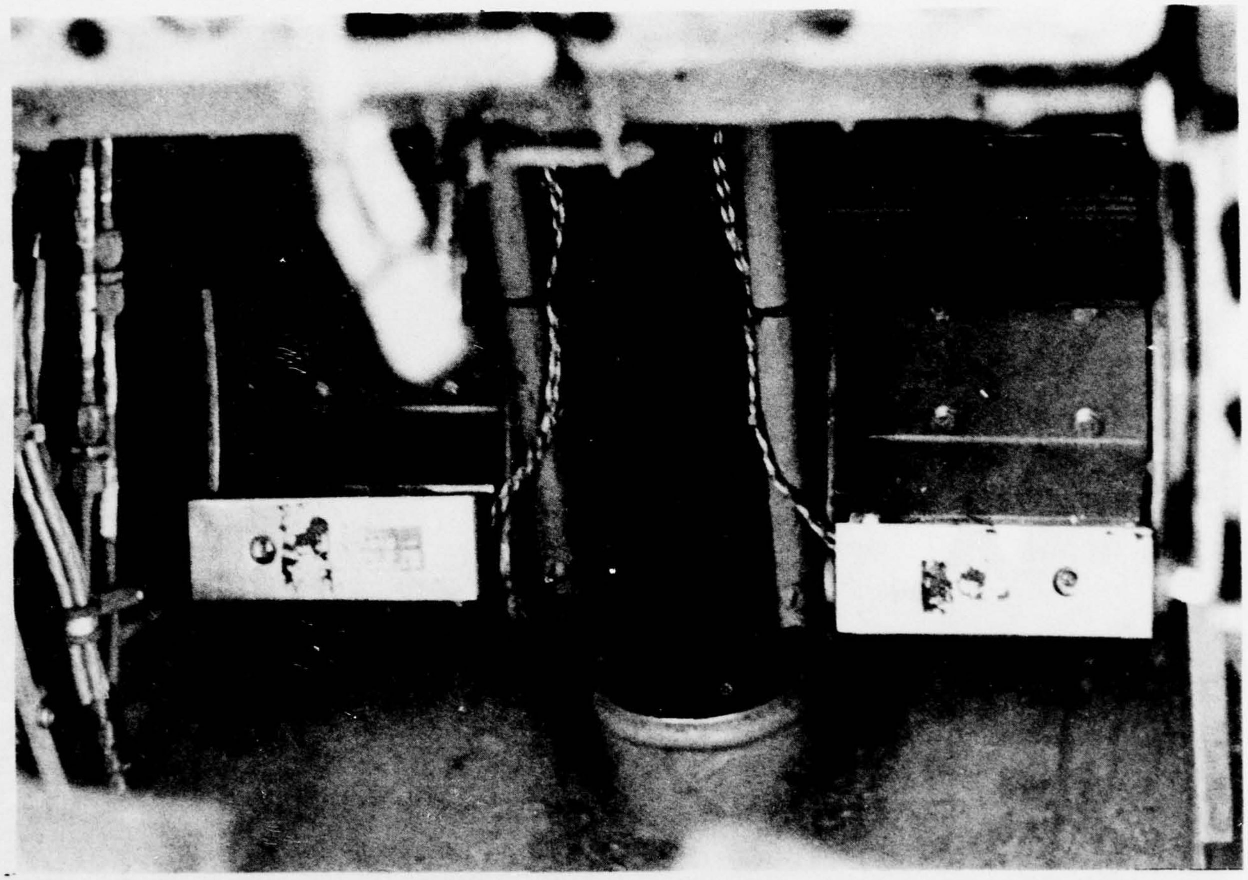


Photo 4. Strain Gaged Rudder Pedals.

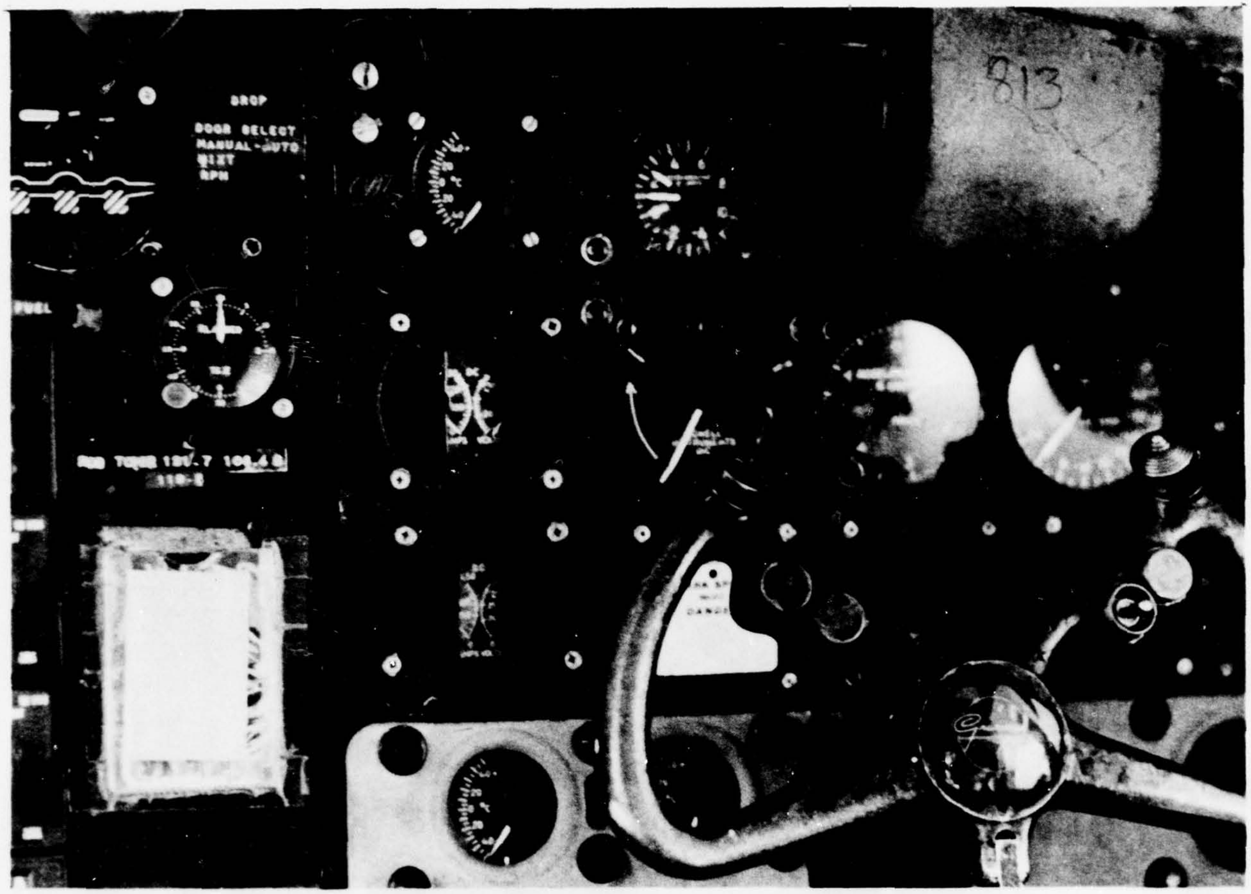


Photo 5. Co-Pilot's Station.

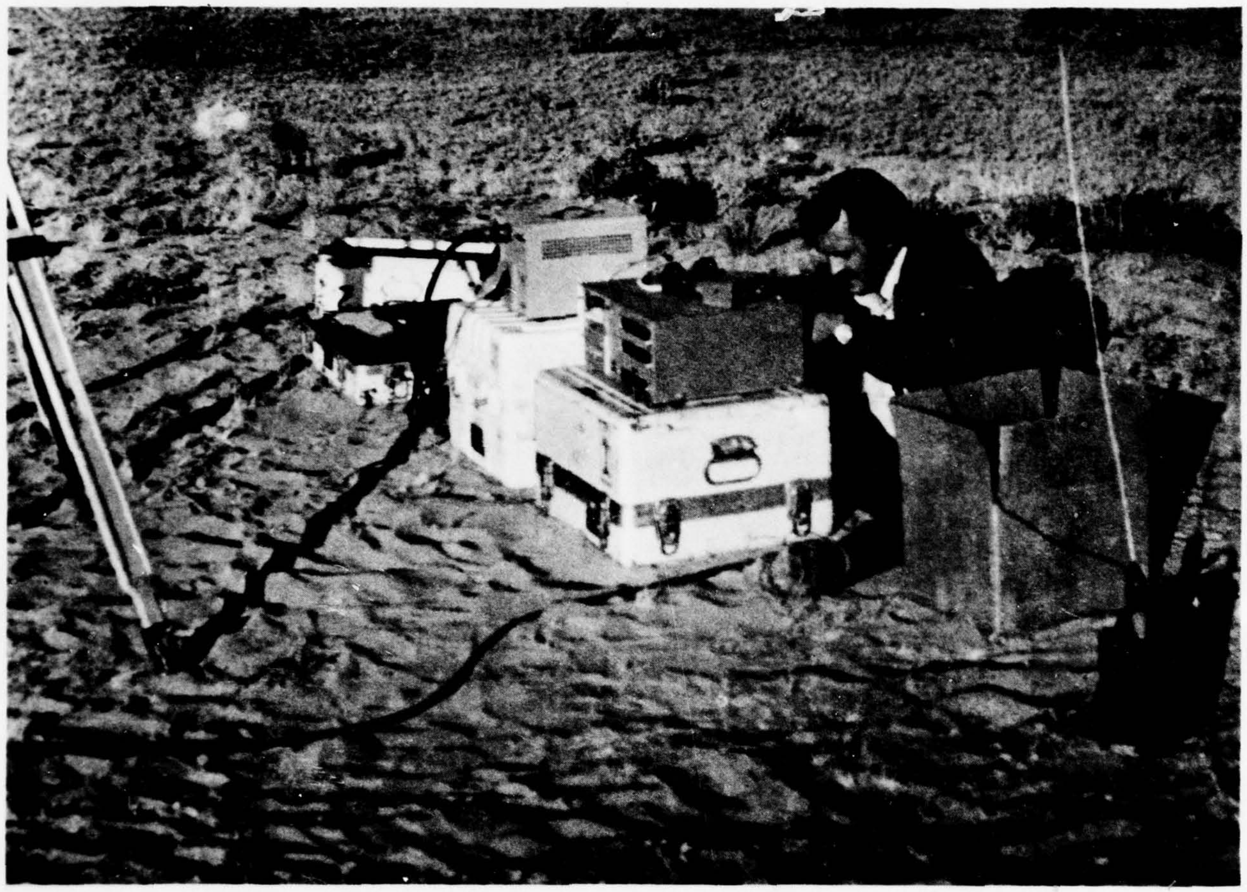


Photo 6. Recording Observation Instrument.

Rate:

- Pitch
- Roll
- Yaw

Center-of-gravity normal acceleration
Engine manifold pressure (both engines)
Engine propeller speed (both engines)
Airspeed
Altitude
Free air temperature
Time of day
Flap position
Pilot event

APPENDIX D. DATA ANALYSIS METHODS

PERFORMANCE

General

1. A limited performance evaluation of the TS-2A airtanker was conducted to define performance changes resulting from the airtanker conversion. All data collected were compared with data from the Phase I report and the NATOPS manual. All tests were flown at the configurations and conditions presented in tables 1 and 2 of the Introduction section of this report. The engine performance chart for the Wright R-1820 engine was used to determine test brake horsepower (BHP_t) (fig. 11A-8 of the NATOPS manual). Brake horsepower standard day (BHP_s) was determined by correcting for variations in manifold pressure and temperature as follows:

$$\Delta BHP = BHP_t \left(\frac{MP_s}{MP_t} - 1\right) + BHP_t \left[\left(\frac{T_t}{T_s}\right)^{0.5} - 1\right] \qquad (1)$$

Where:

$\Delta BHP = BHP_t - BHP_s$

MP_s = Manifold pressure standard (in. Hg)

MP_t = Manifold pressure test (in. Hg)

T_t = Temperature test (°K)

T_s = Temperature standard (°K)

Corresponding values of mean net propeller thrust (Fp) were then calculated at an average true airspeed between velocity at liftoff and 50 feet.

$Fp = \Delta BHP/V_{T(AVERAGE)}$

Takeoff Performance

2. Takeoff performance tests were conducted at two pressure altitudes (2300 and 6200 feet) to determine ground roll distance and air distance to clear a 50-foot obstacle. Quantitative data were recorded on board the aircraft and also at a ground station, using Recording Optical Instruments (ROI). ROI ground speed data were used to determine true airspeed at liftoff and 50 feet. The pitot-static position error (ΔV_{pc}) was computed from the following relationship:

$$\Delta V_{pc} = (V_{Tt} \times \sqrt{\sigma_t}\ ) - Vi$$

and calibrated airspeed (V_c) was obtained by correcting the instrument corrected indicated airspeed for position error (ΔV_{pc}).

$$V_c = Vi + \Delta V_{pc}$$

Where:

V_{Tt} = Test true airspeed (knots)

Vi = Indicated airspeed (instrument corrected)

ΔV_{pc} = Pitot-static position error (knots)

3. Takeoff data were corrected to standard conditions. A correction for winds less than 5 knots, using equations 2 and 3, was used.

$$Sg_w = Sg \left(1 + \frac{V_w}{V_{TO}}\right)^{1.85} \quad (2)$$

$$Sa_w = Sa + V_w T \quad (3)$$

Where:

Sg = Ground distance (ft)

Sa = Air distance (ft)

V_w = Wind velocity (ft/sec)

V_{TO} = Velocity at takeoff (ft/sec)

T = Time from liftoff to 50 feet (sec)

Sg_w = Ground distance corrected for wind (ft)

Sa_w = Air distance corrected for wind (ft)

Corrections for runway slope were made with the following equation:

$$Sg_{SL} = \frac{}{1 + \frac{2g\ Sg}{V_{TO}^{2}} \sin\theta} \quad (4)$$

Where:

Sg_{SL} = Ground distance corrected for slope (ft)

θ = Runway slope (positive uphill in degrees)

g = Acceleration due to gravity - 32.1741 ft/sec^2

The combined equations for thrust, weight, and density corrections are shown below.

$$\frac{Sg_s}{Sg_t} = \frac{\frac{W_s}{W_t}\ \frac{\sigma_t}{\sigma_s}}{\frac{2g\ Sg_t}{W_t\ V_{TO_t}^2}\ \left(\frac{W_t}{W_s}\ (F_{P_s} - F_{P_t})\right) + 1} \quad \text{(Ground Distance) (5)}$$

using

$$h_{vt} = \frac{V_{50}^2 - V_{TO}^2}{2g}$$

$$\frac{Sa_s}{Sa_t} = \frac{\left[\frac{W_s}{W_t}\ \frac{\sigma_t}{\sigma_s}\ h_{vt}\right] + 50}{(h_{v_t} + 50) + \frac{Sa_t\ F_{P_s}}{W_s} - \frac{Sa_t\ F_{P_t}}{W_t}} \quad \text{(Air Distance) (6)}$$

Where:

Sg = Ground distance (ft)

Sa = Air distance (ft)

W_s = Standard gross weight (pounds)

W_t = Test gross weight (pounds)

θ_t = Test density ratio

θ_s = Standard density ratio

g = Acceleration due to gravity (ft/sec^2)

V_{TO} = Velocity at lift-off (ft/sec)

LBF/ft = Pound force

AGL = Above ground level

V_{50} = Velocity at 50 feet AGL (ft/sec)

Fp = Mean net propellor thrust

4. Comparing aircraft takeoff performance data at different combinations of takeoff weights, altitudes, and airspeeds can be conveniently done by dealing in aircraft total energy (E). At any given time the total energy (E) possessed by an aircraft is the sum of the potential (PE) and kinetic (KE) energies. Total aircraft energy is calculated using equation 7. Total ground distance to lift-off and to 50 feet AGL was plotted against corresponding total aircraft energy. These plots are shown below.

$$E = \frac{1}{2} \frac{W_s}{g} V_{T_s}^{2} (1.689)^2 + W_s h \tag{7}$$

$$V_{T_s} = \left(\frac{W_s}{W_t} \frac{\sigma_t}{\sigma_s}\right)^{1/2} V_{T_t} \tag{8}$$

Where:

V_{T_s} = Standard true airspeed (knots)

V_{T_t} = Test true airspeed (knots)

E = Aircraft energy level (LBF/FT)

h = Aircraft height (feet)

Figure A.

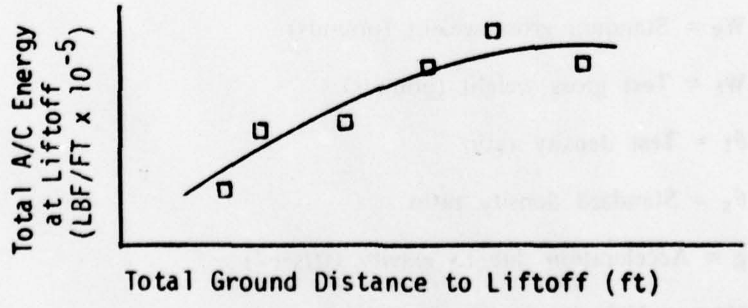
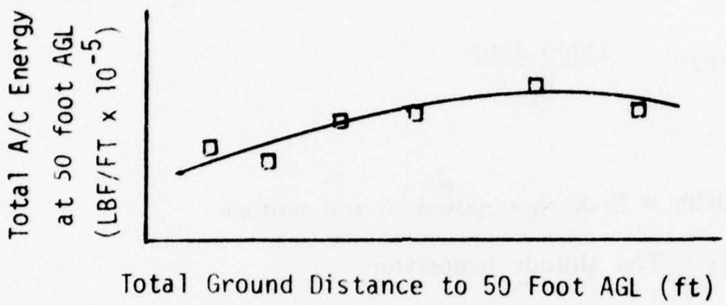


Figure B.



5. Curves similar to those shown above were used for selecting adjusted values of aircraft energy levels at given ground distances to lift-off and 50 feet AGL. Equation 9 was then used to calculate a new calibrated airspeed using the adjusted aircraft energy level. These calculated values of calibrated airspeed at given ground distances define the faired curves shown in figures 1 and 2, appendix E.

$$V_{CAL} = 0.5921\left[(E - W_s h)\frac{2g}{W_s}\right]^{1/2}(\sigma_s)^{1/2} \qquad (9)$$

Climb Performance

6. Climb performance was investigated using the sawtooth climb method. The sawtooth climbs were performed through pressure altitudes of 6000 and 8000 feet. A constant manifold pressure and constant propeller speed were used during climb performance. The recorded rate of climb (R/C) was corrected for temperature, power and gross weight variations using the pressure altitude method, as follows:

$$R/C_{CORR} + R/C_{TAPELINE} + \Delta R/C_1 + \Delta R/C_2 \qquad (10)$$

TAPELINE R/C at test day true airspeed is shown in equation 11.

$$R/C_{TAPELINE} = dh/dt\ (T_t + 273.15/T_s + 273.15)^{1/2} \qquad (11)$$

Where:

dh/dt = recorded R/C

T_t = Test altitude temperature (°C)

T_S = Test altitude standard day temperature (°C)

The rate-of-climb correction for power variation with temperature during a climb at constant manifold pressure and propeller speed is represented by equation 12.

$$\Delta R/C_1 = \frac{33000\ \Delta BHP}{W_t} \qquad (12)$$

Where:

BHP_t = Brake horsepower at test altitude

T_t = Test altitude temperature

T_s = Test altitude standard day temperature

W_t = Test gross weight

The R/C correction for gross weight variation is given by equation 13.

$$\Delta R/C_2 = (R/C_{TAPELINE} + \Delta R/C_1 \left(\frac{W_t - W_s}{W_t}\right) \qquad (13)$$

Where:

W_s = Gross weight standard

<u>Acceleration/Stop Performance</u>

7. Acceleration and deceleration tests were performed to determine the total field length required to safely stop the aircraft following an engine failure. Total distance for power acceleration and braking deceleration was recorded from a ground station using an ROI. Data were corrected to standard day using equation 5 to represent the acceleration period and equation 14 to correct the deceleration distance.

$$Sg_s = Sg_t \left(\frac{W_s}{W_t}\right)^2 \left(\frac{\sigma_t}{\sigma_s}\right) \qquad (14)$$

Landing Performance

8. Short field landing performance tests were conducted to determine ground roll distance and total distance from over a 50-foot obstacle. Landing tests were performed at 19,000 pounds. Quantitative data were recorded on board the aircraft and from a ground station utilizing an ROI. Recorded landing distances were corrected to standard day utilizing equation 5 from ground roll corrections and equation 15 for air distance corrections (ref 6, app A).

$$S_{a_s} = S_{a_t} \left(\frac{W_t}{W_s}\right)^2 + \frac{h_v}{h_v + 50} \times \left(\frac{\sigma_t}{\sigma_s}\right) \frac{h_v}{h_v + 50} \qquad (15)$$

Where:

$$h_V = \frac{V_{50}{}^2 - V_{TO}{}^2}{2g}$$

HANDLING QUALITIES

General

9. The handling qualities evaluation was conducted to determine the effect on handling qualities of the airtanker conversion. All tests were flown at the conditions shown in tables 1 and 2 of the introduction. Handling qualities test data are shown in appendix E.

Mission Maneuvering Characteristics

10. The retardant delivery evaluation was to determine a safe mission envelope for the operational pilot. This was accomplished by using a buildup in airspeed and load factor during retardant delivery. These retardant deliveries were made using a matrix of three flap settings, landing gear up and down, and full and half salvo (2x2) sequencing. The aircraft was stabilized at the aim altitude and airspeed prior to initiation of the retardant delivery and time histories of all the measured parameters were recorded on magnetic tape. Three flights utilizing actual fire retardant were performed to determine if there was any difference between the effects of water and retardant on normal acceleration.

Weight and Balance

11. The aircraft weight and cg was determined utilizing the Edwards AFB, California, weight and balance facility prior to the start of flight testing. The aircraft was weighed empty, full of water (800 gals) and full of water and fuel. A fuel

cell calibration was also performed and the fuel cell capacity was recorded at 532 gallons. The aircraft cg was at FS 216.4 with test instrumentation installed less fuel and water, FS 217.05 full water, and FS 217.21 water and fuel.

Airspeed Calibration

12. The ship's standard pitot-static system was calibrated using the pace vehicle method to determine the airspeed position error (fig. 34, app E). Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}).

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \tag{16}$$

13. True airspeeds (V_T) were determined from the test altitude air density ratio (θ) and equivalent airspeed (V_e), as follows:

$$V_T = \frac{V_e}{\sqrt{\sigma}}$$

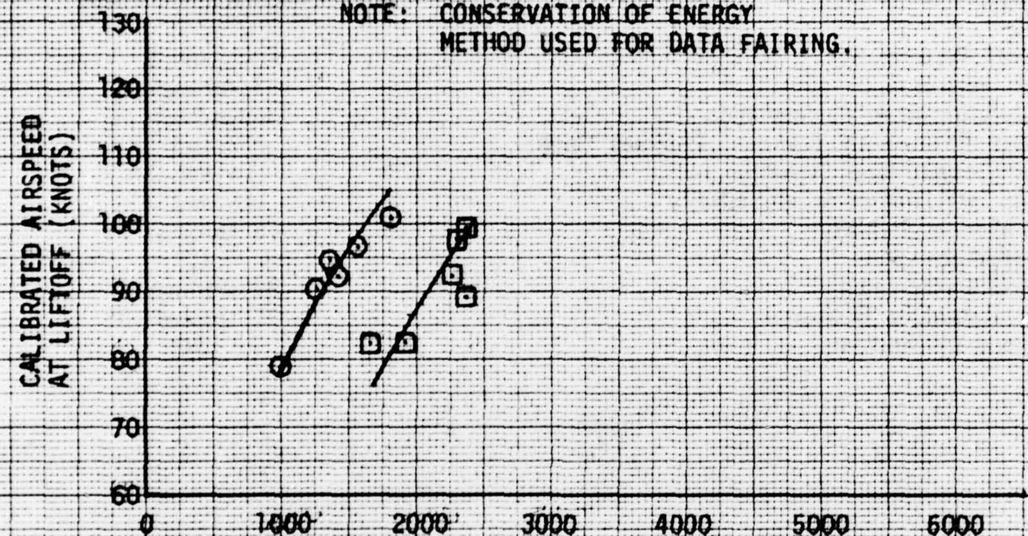
APPENDIX E. TEST DATA

INDEX

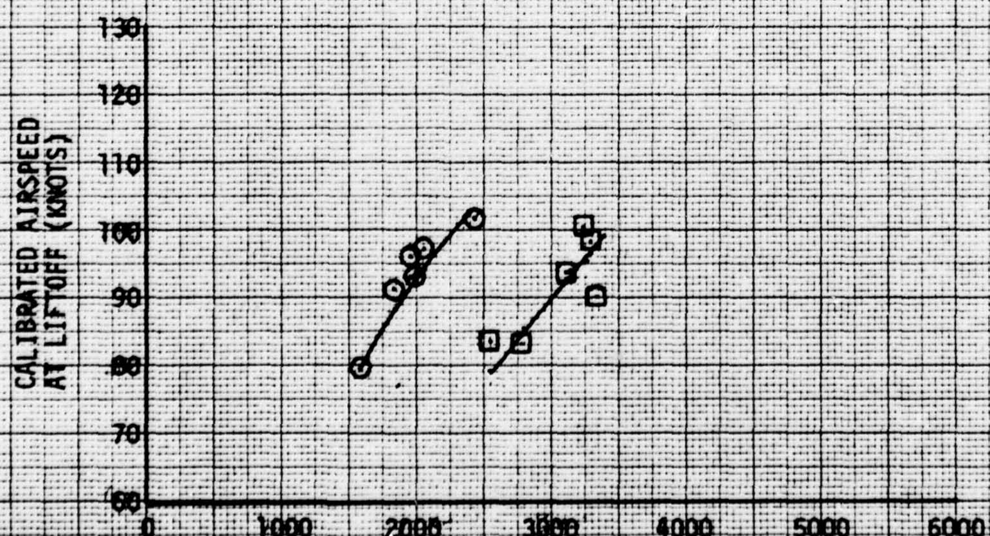
FIGURE 7
TAKEOFF PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
⊙	19220	215.8	2520	13.0	2800	49
⊡	19320	215.8	6140	8.5	2800	41

NOTE: CONSERVATION OF ENERGY METHOD USED FOR DATA FAIRING.



RUNWAY DISTANCE (FEET)

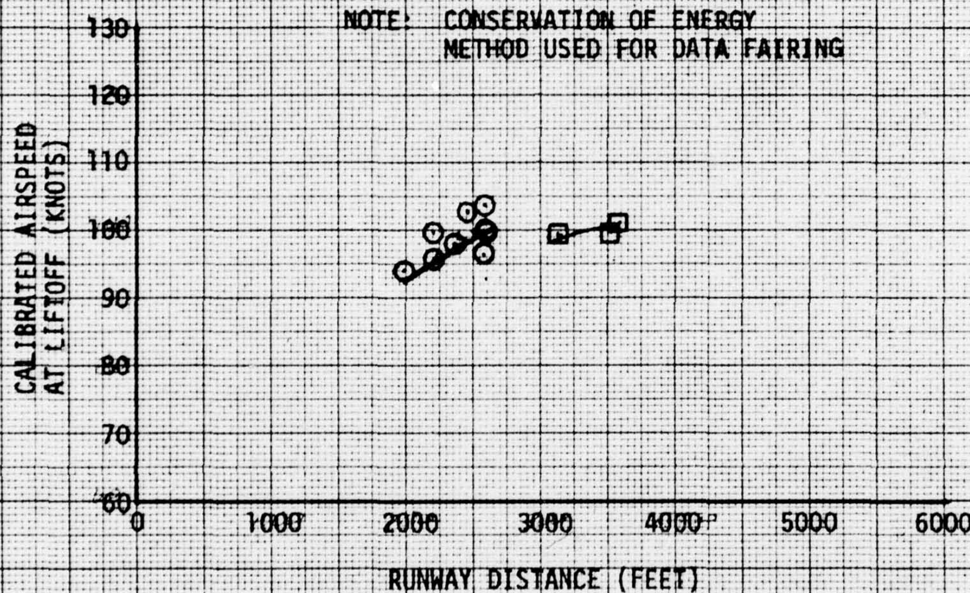


TOTAL DISTANCE TO CLEAR 50 FOOT OBSTACLE (FEET)

FIGURE 2
TAKEOFF PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
⊙	26320	215.6	2440	12.5	2800	49
⊡	26380	215.6	5460	0.5	2800	41

NOTE: CONSERVATION OF ENERGY METHOD USED FOR DATA FAIRING



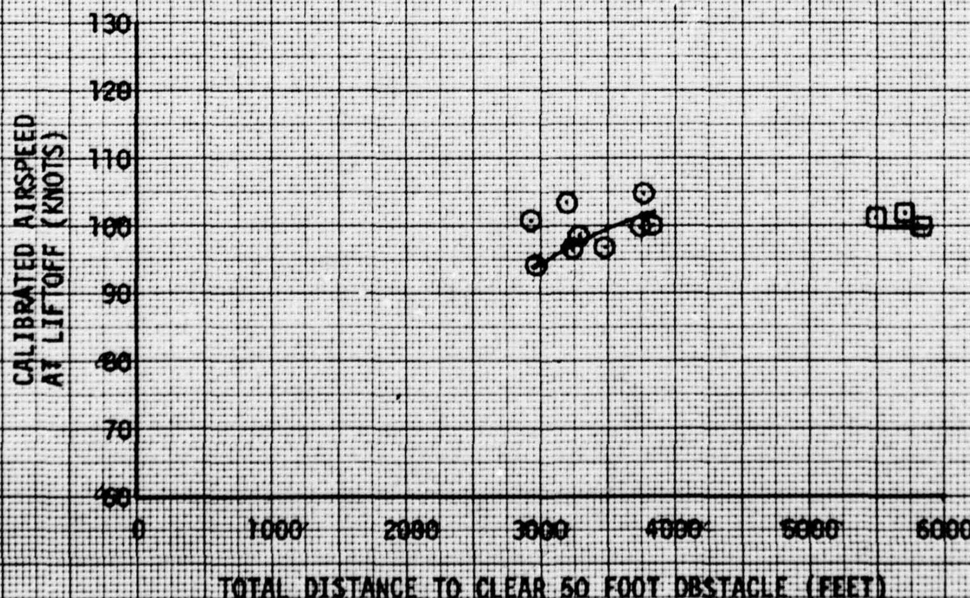


FIGURE 3
CLIMB PERFORMANCE
TS-2A USFS S/N N404DF

GROSS WEIGHT = 25800 LBS
LONG C.G. LOCATION = 215.8 IN
ALTITUDE = 6000 FEET (STA. DAY)
PROPELLER SPEED = 2300 RPM
POWER SETTING = MAX CONTINUOUS (35 IN/HG)

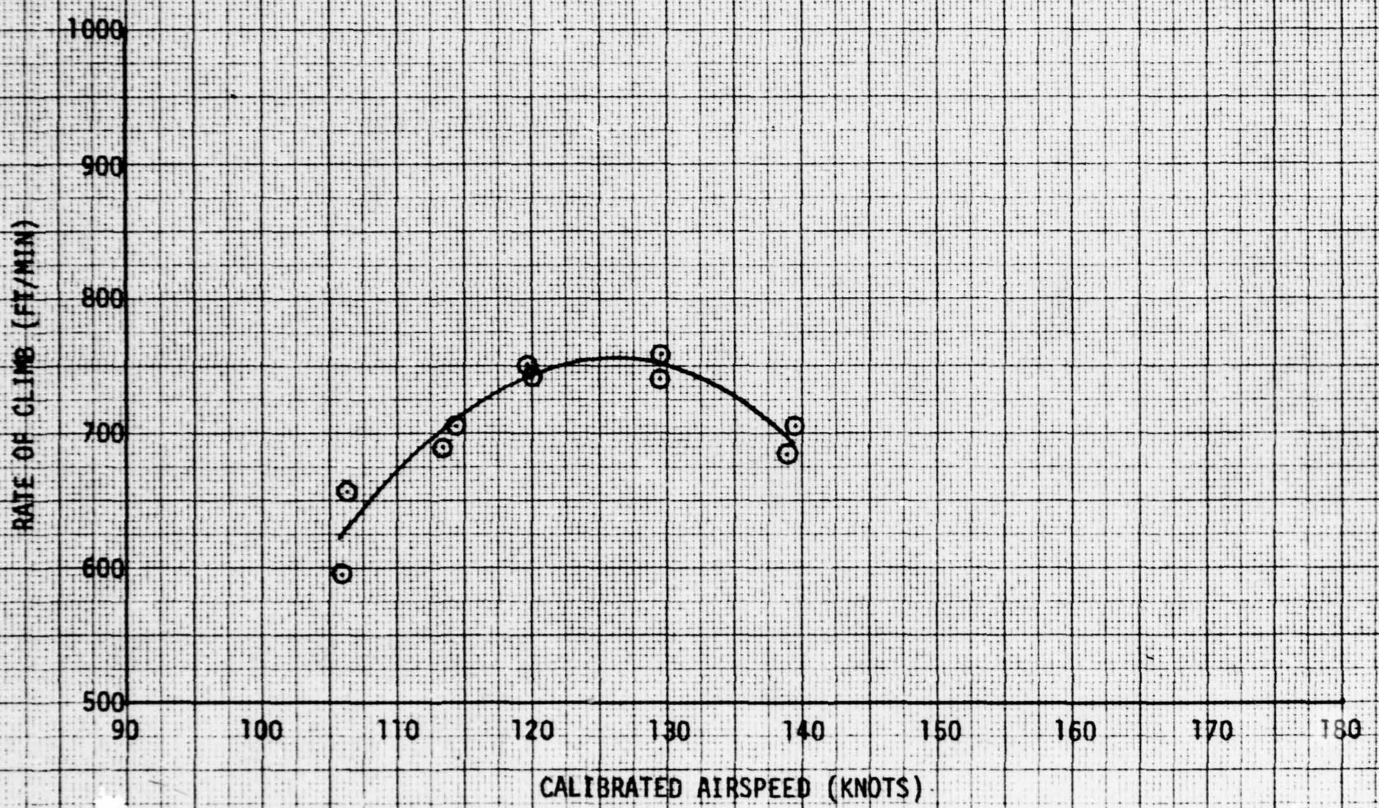


FIGURE 4
CLIMB PERFORMANCE
TS-2A USFS S/N N404DF

GROSS WEIGHT = 18620 LBS
LONG C.G. LOCATION = 216.2 IN
ALTITUDE = 6000 FEET (STA. DAY)
PROPELLER SPEED = 2300 RPM
POWER SETTING = MAX CONTINUOUS (35 IN/HG)

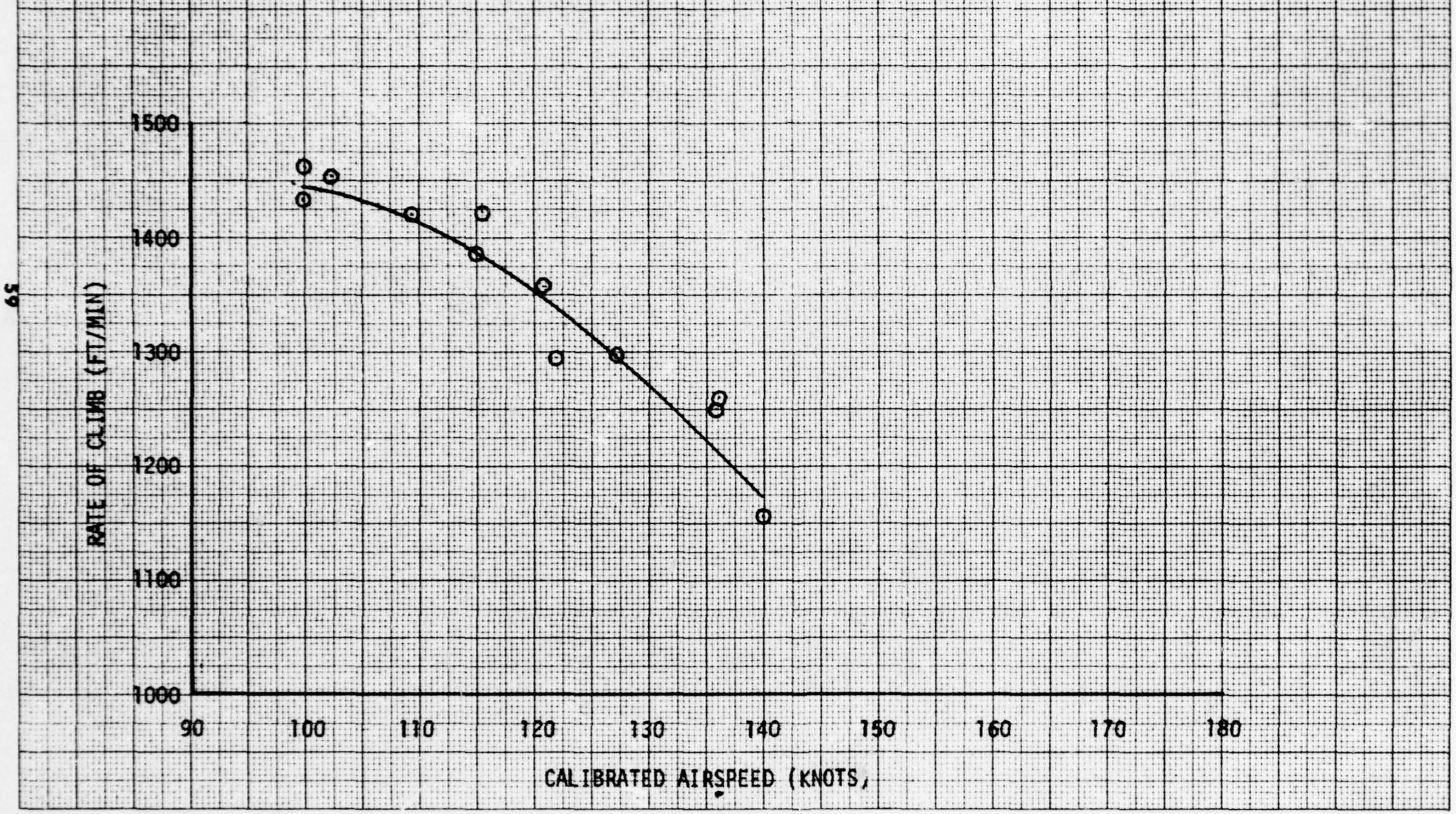


FIGURE 5
SINGLE ENGINE CLIMB PERFORMANCE
TS-2A USFS S/N N404DF

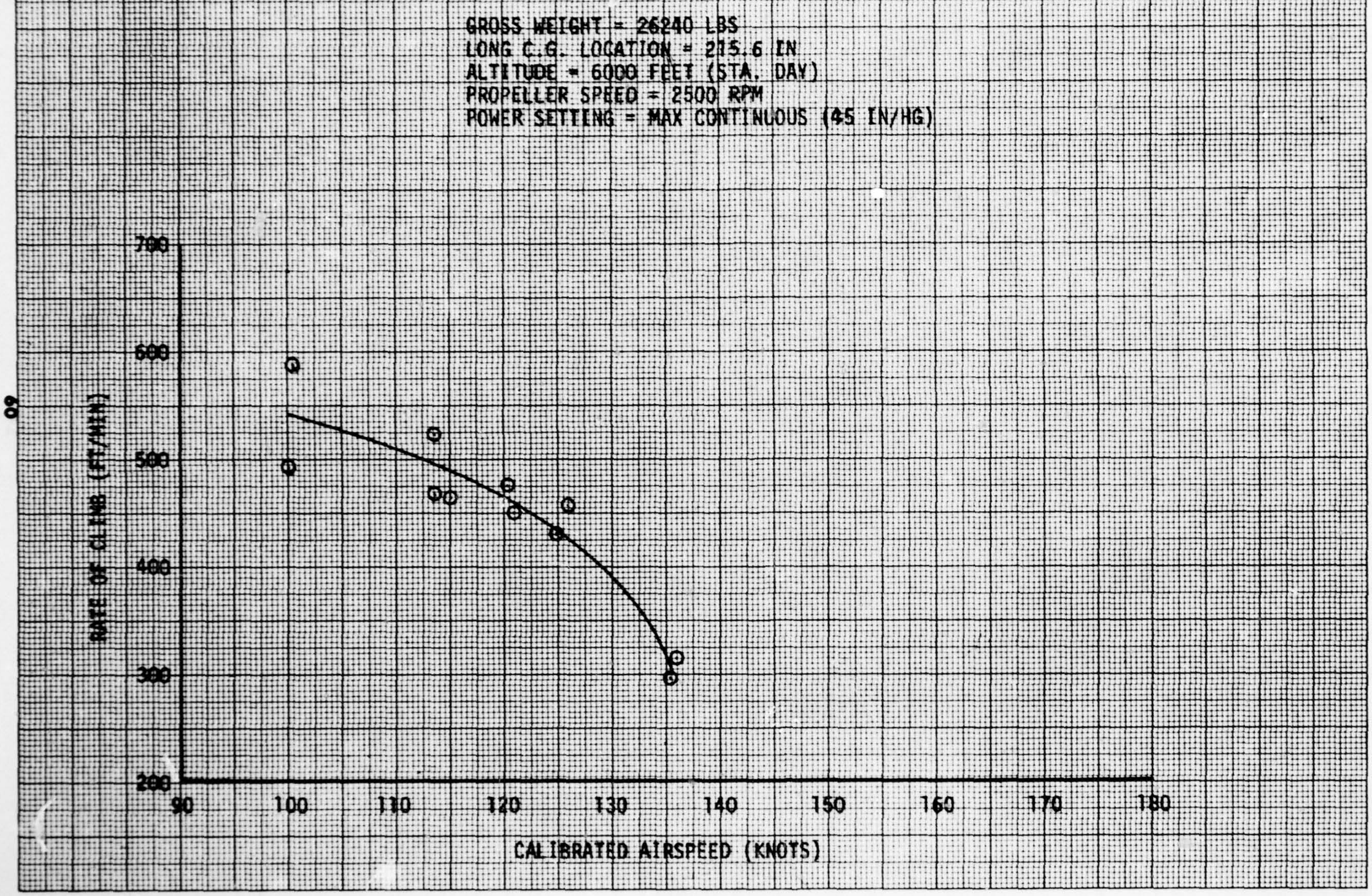
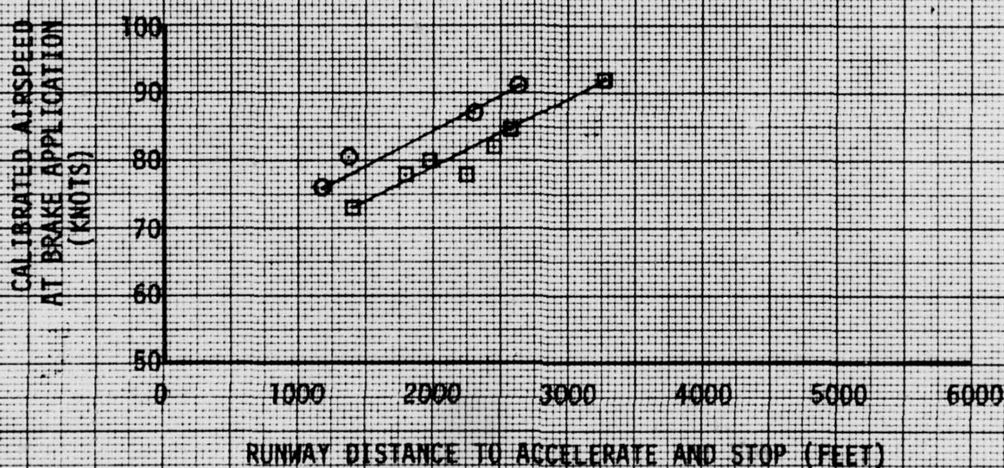


FIGURE 6

ACCELERATE/STOP PERFORMANCE

TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19320	216.8	2100	10.5	2800	49
□	19300	215.6	5640	3.5	2800	41



SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	26360	215.6	2180	10.5	2800	49
□	26360	215.6	5380	0	2800	41

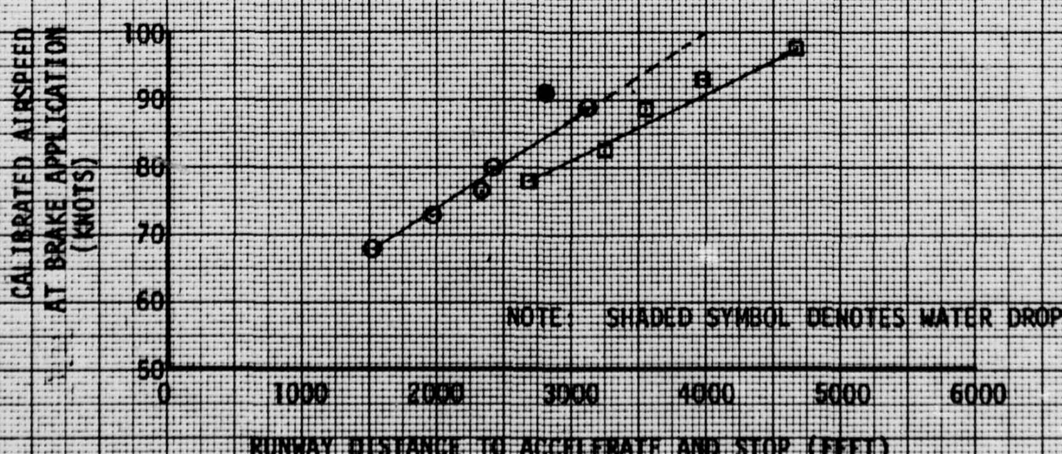
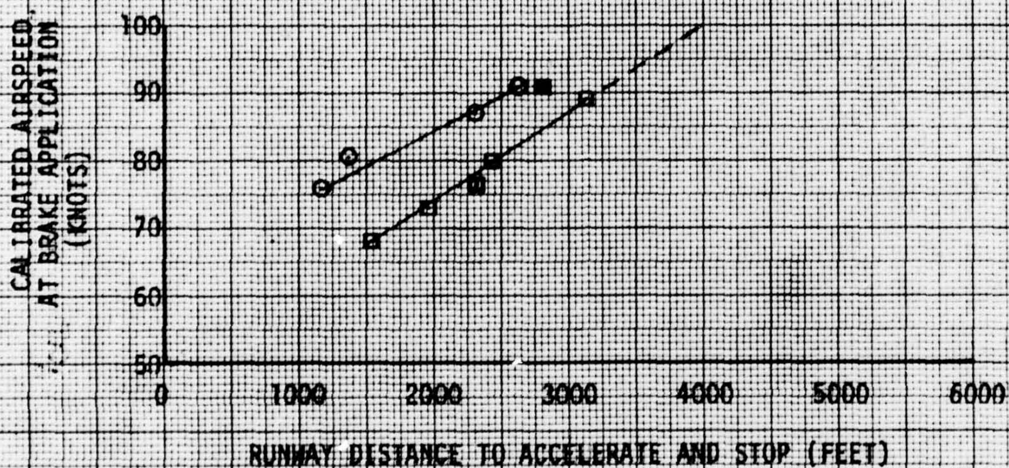


FIGURE 7
ACCELERATE/STOP PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19320	215.8	2100	10.5	2800	49
□	26360	218.6	2180	10.5	2800	49

NOTE: SHADED SYMBOL DENOTES WATER DROP



SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19300	215.8	5640	3.5	2800	41
□	26360	215.6	5380	0	2800	41

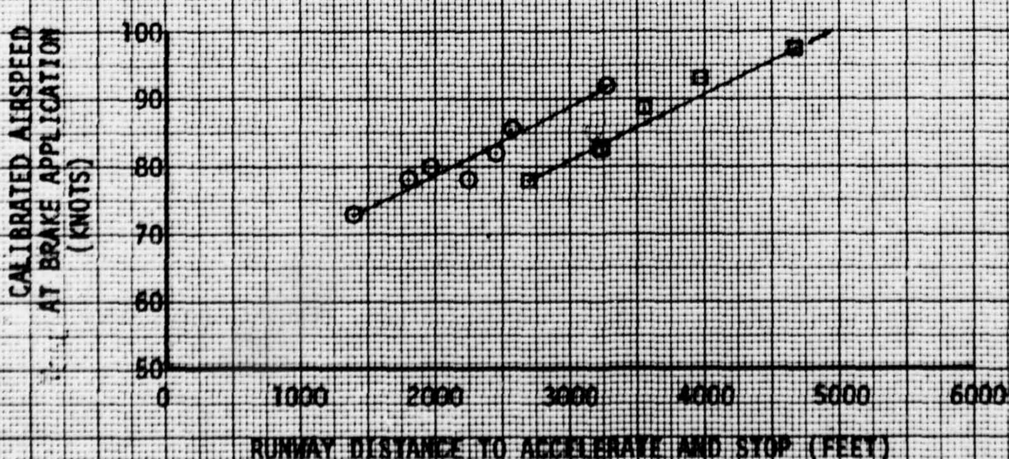
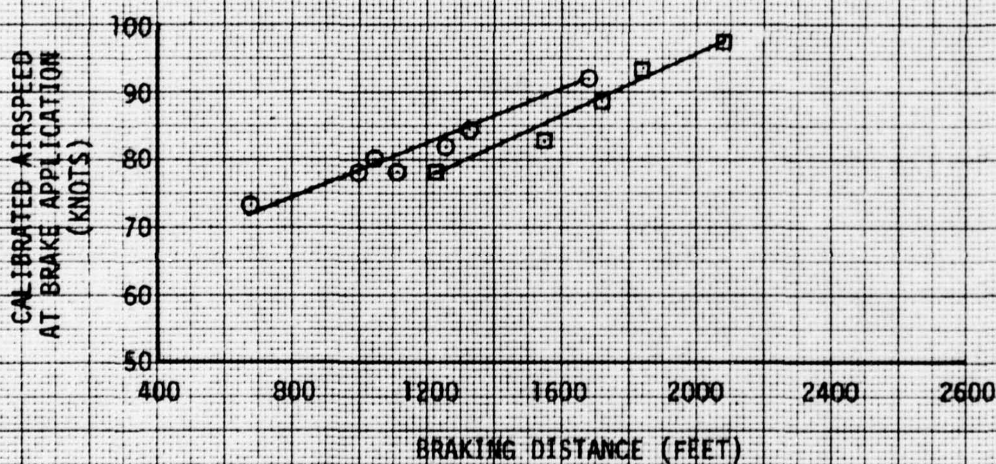


FIGURE 8
ACCELERATE/STOP PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19300	215.8	5640	8.5	2800	41
□	26380	215.6	5380	0	2800	41



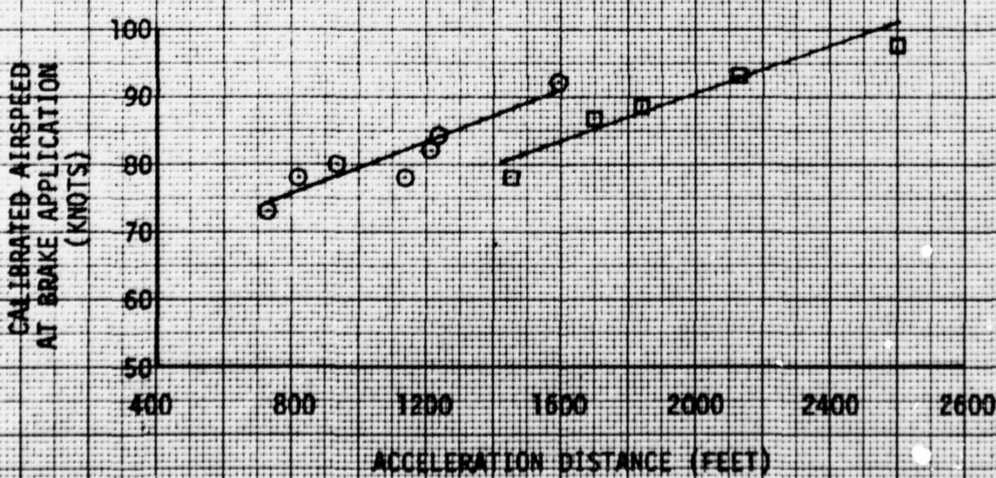
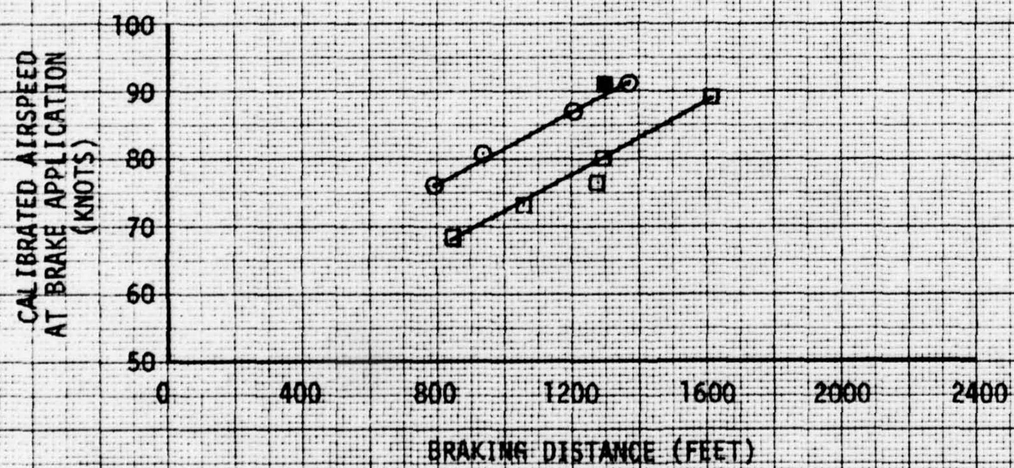


FIGURE 9
ACCELERATE/STOP PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19320	216.8	2100	10.5	2800	49
□	26360	216.6	2180	10.5	2800	49

NOTE: SHADED SYMBOL DENOTES WATER DROP



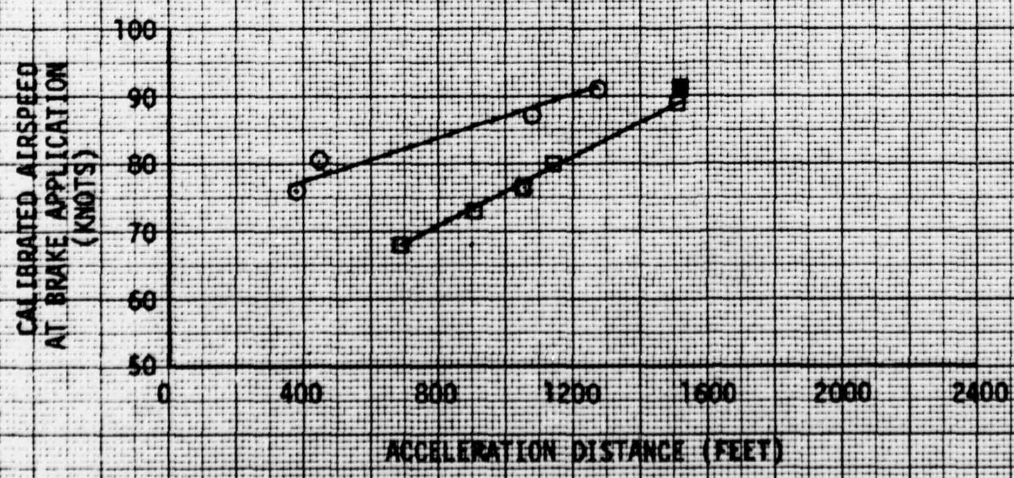


FIGURE 10
ACCELERATE/STOP PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
○	19320	215.8	2100	10.5	2800	49
□	19300	215.8	5640	3.5	2800	41

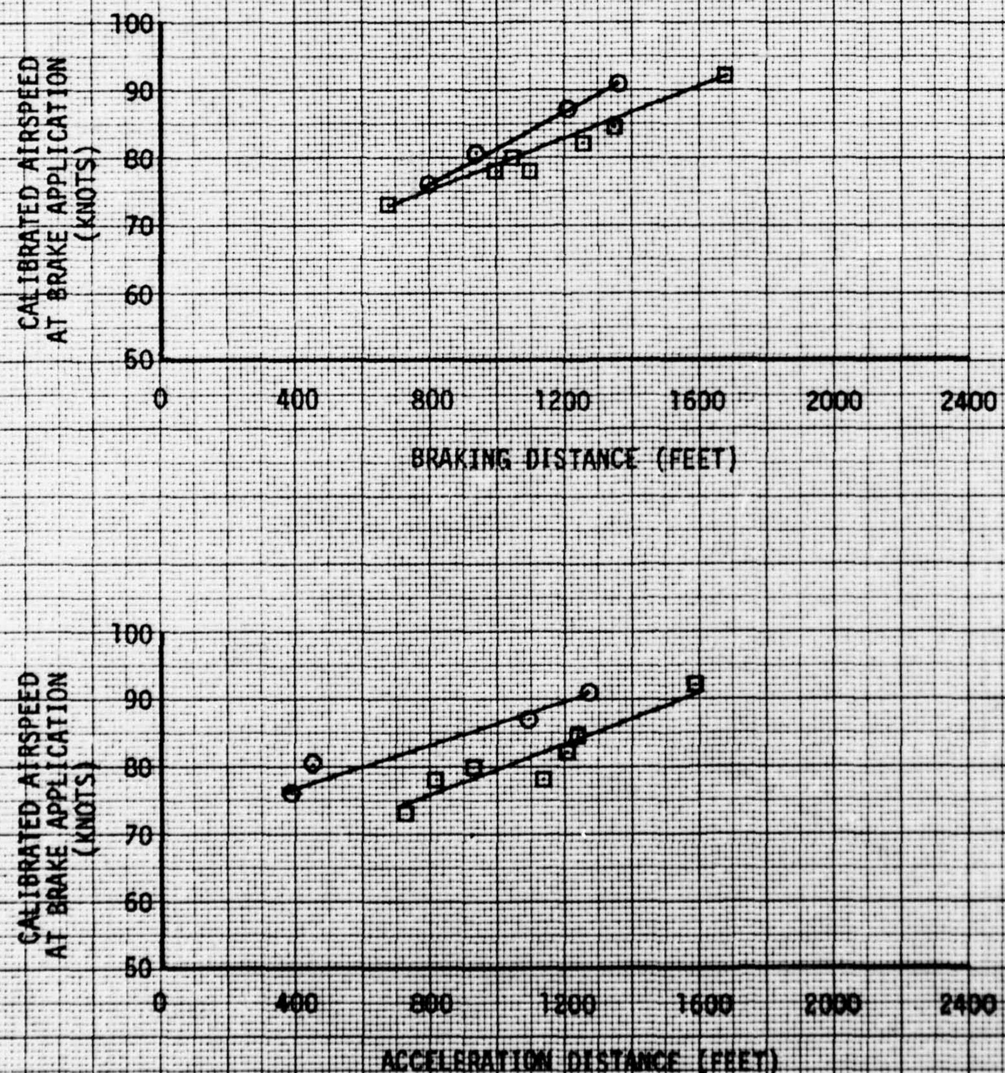
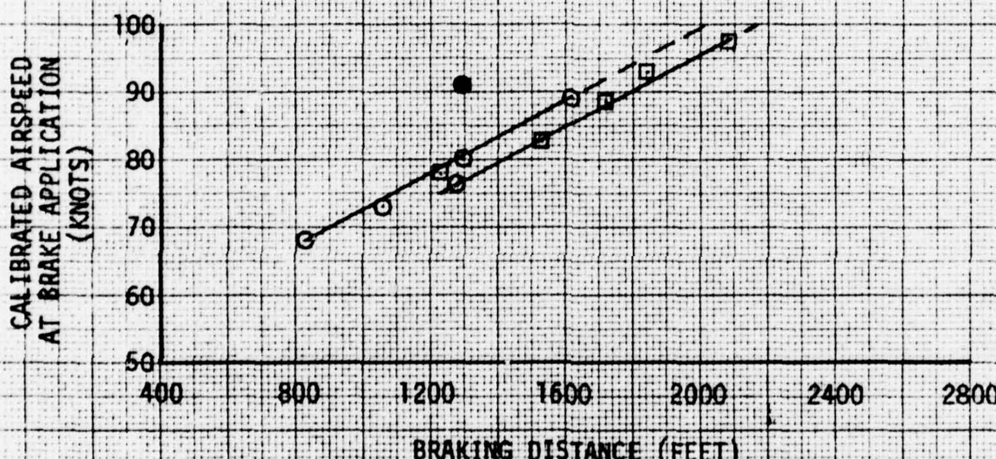


FIGURE 11
ACCELERATE/STOP PERFORMANCE
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	MANIFOLD PRESSURE (IN/HG)
O	26360	215.6	2180	10.5	2800	49
□	26360	215.6	5380	0	2800	41

NOTE: SHADED SYMBOL DENOTES WATER DROP



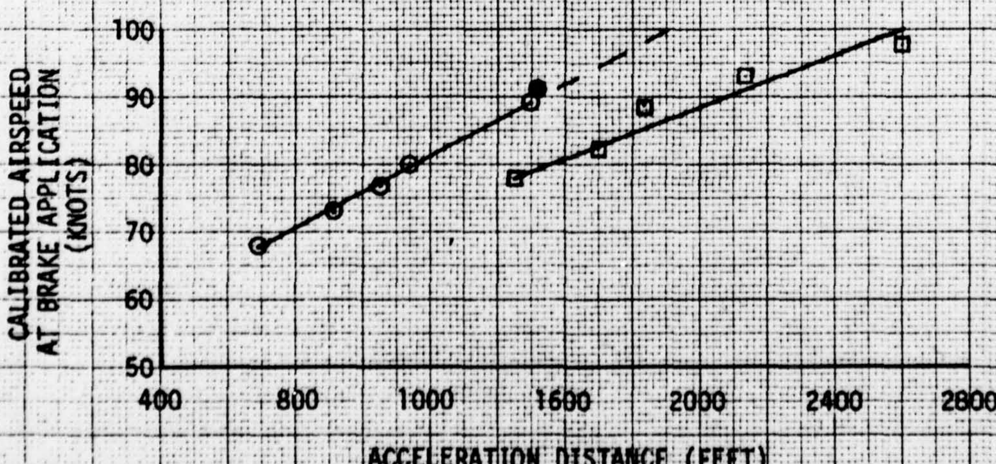


FIGURE 12
CONTROL POSITION IN TRIMMED FORWARD FLIGHT
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
○	18290	215.1	7560	12.5	2300	CLEAN	LEVEL FLT
□	18980	215.9	8960	10.0	2500	LANDING	DESCENT (500 FPM)
△	18980	215.9	7920	13.5	2300	CLEAN	CLIMB (500 FPM)

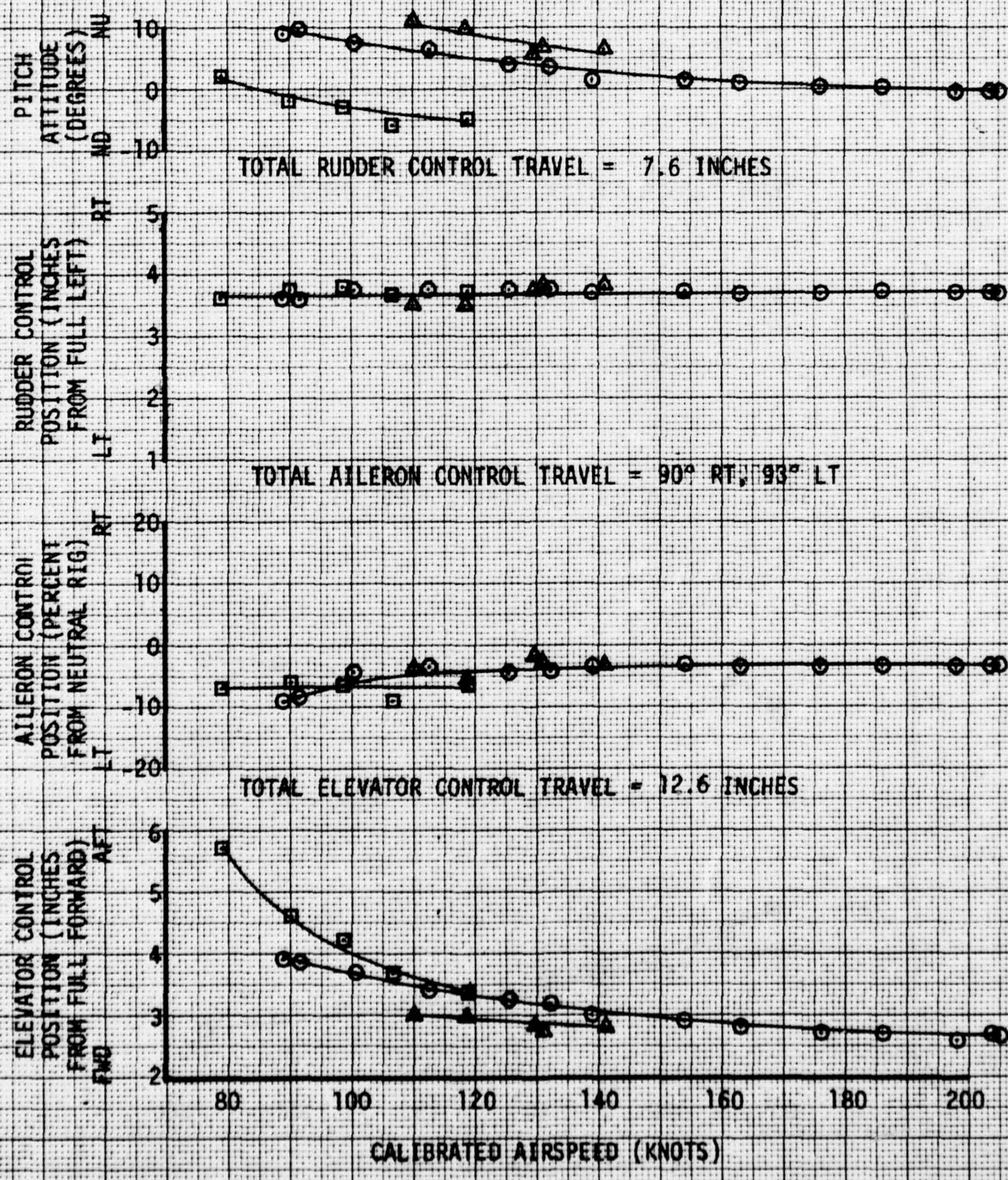


FIGURE 13
CONTROL POSITION IN TRIMMED FORWARD FLIGHT
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
○	26310	216.6	7800	20.0	2300	CLEAN	LEVEL FLT
□	26080	215.7	7640	18.9	2600	CLEAN	DESCENT (500 FPM)
△	26090	215.9	7700	19.0	2300	CLEAN	CLIMB (500 FPM)

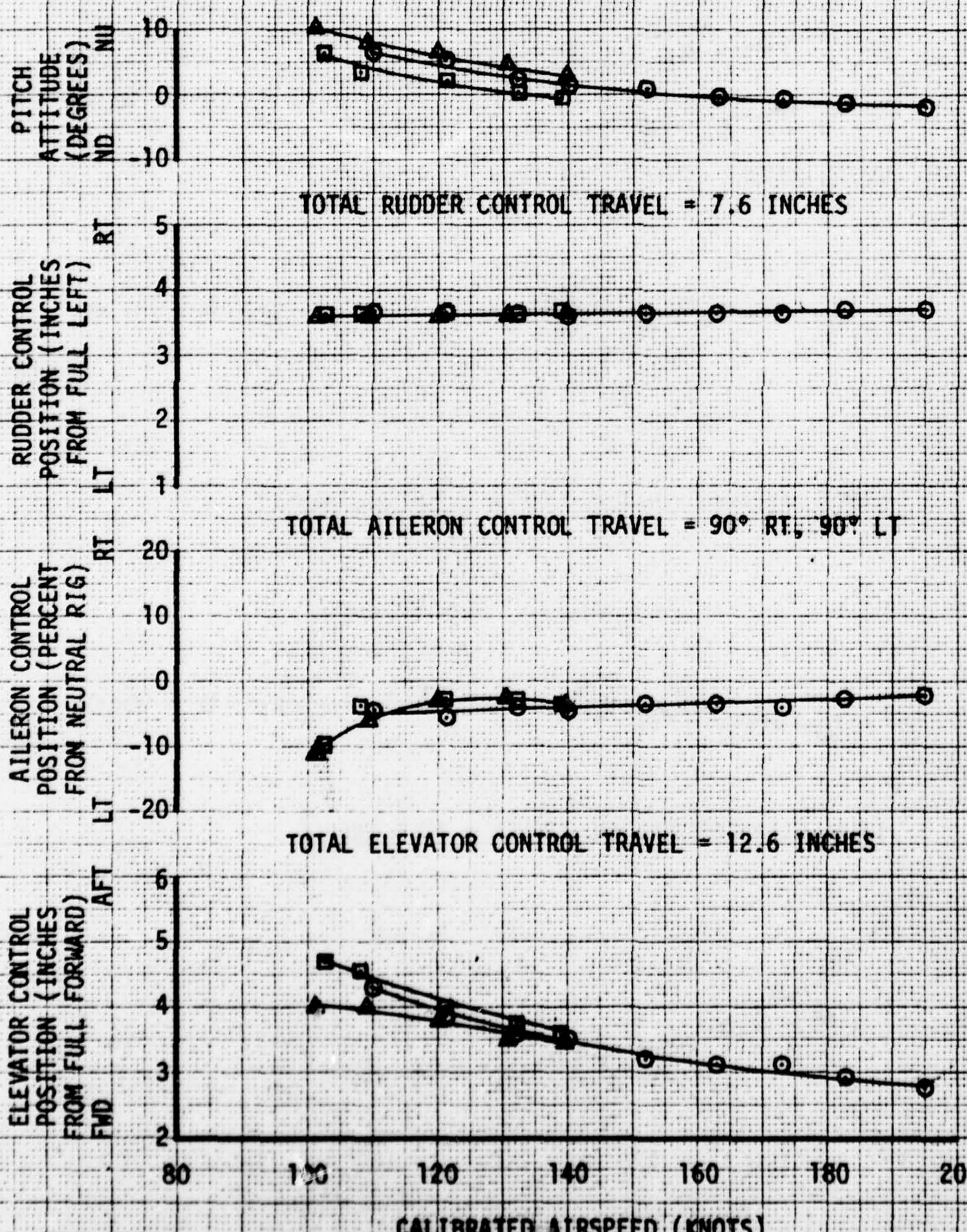


FIGURE 14
STATIC LONGITUDINAL STABILITY
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
○	18880	215.9	8150	14.0	2300	CLEAN	LEVEL FLT

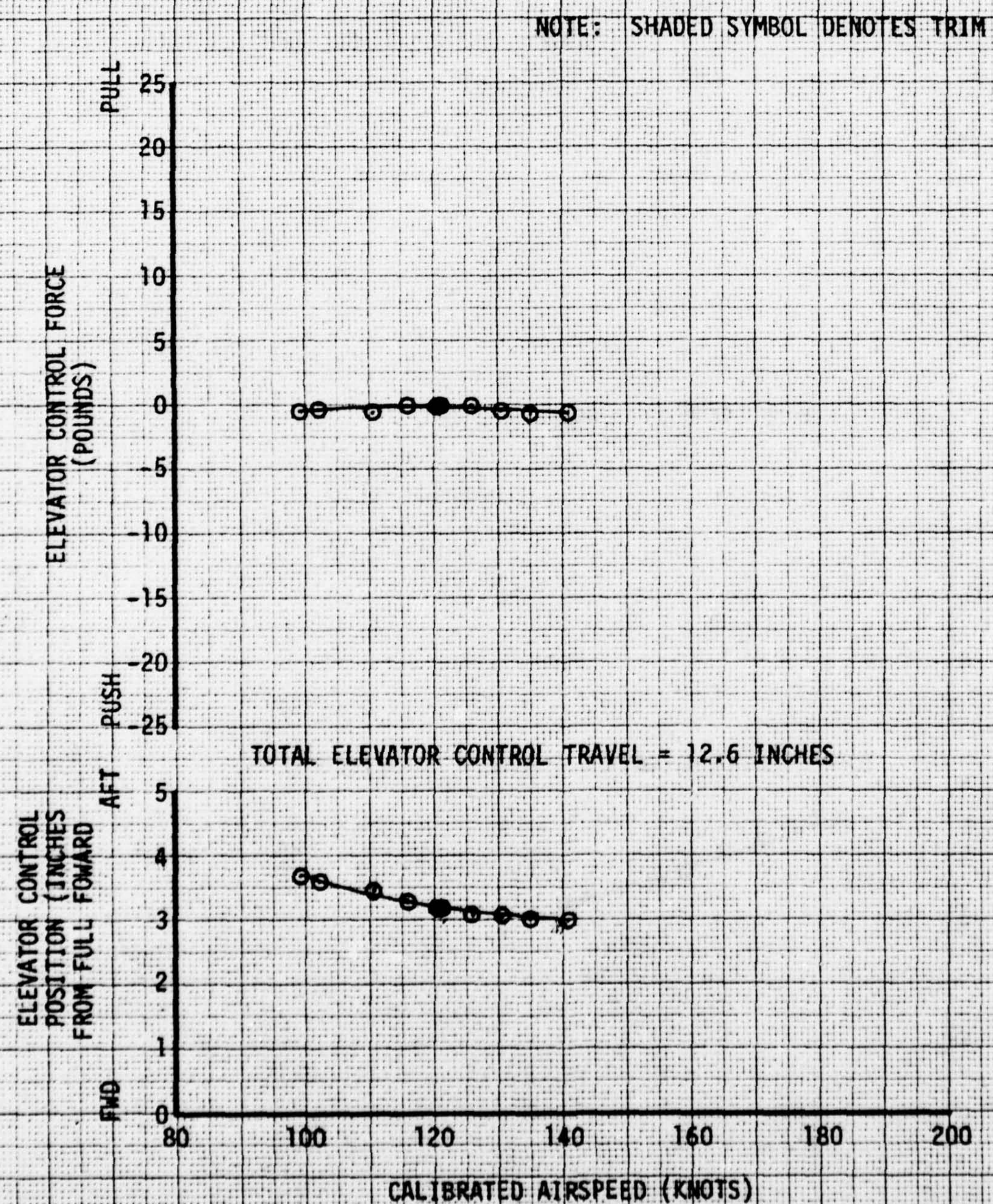


FIGURE 15
STATIC LONGITUDINAL STABILITY
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
O	25900	215.8	7720	18.5	2300	CLEAN	LEVEL FLT

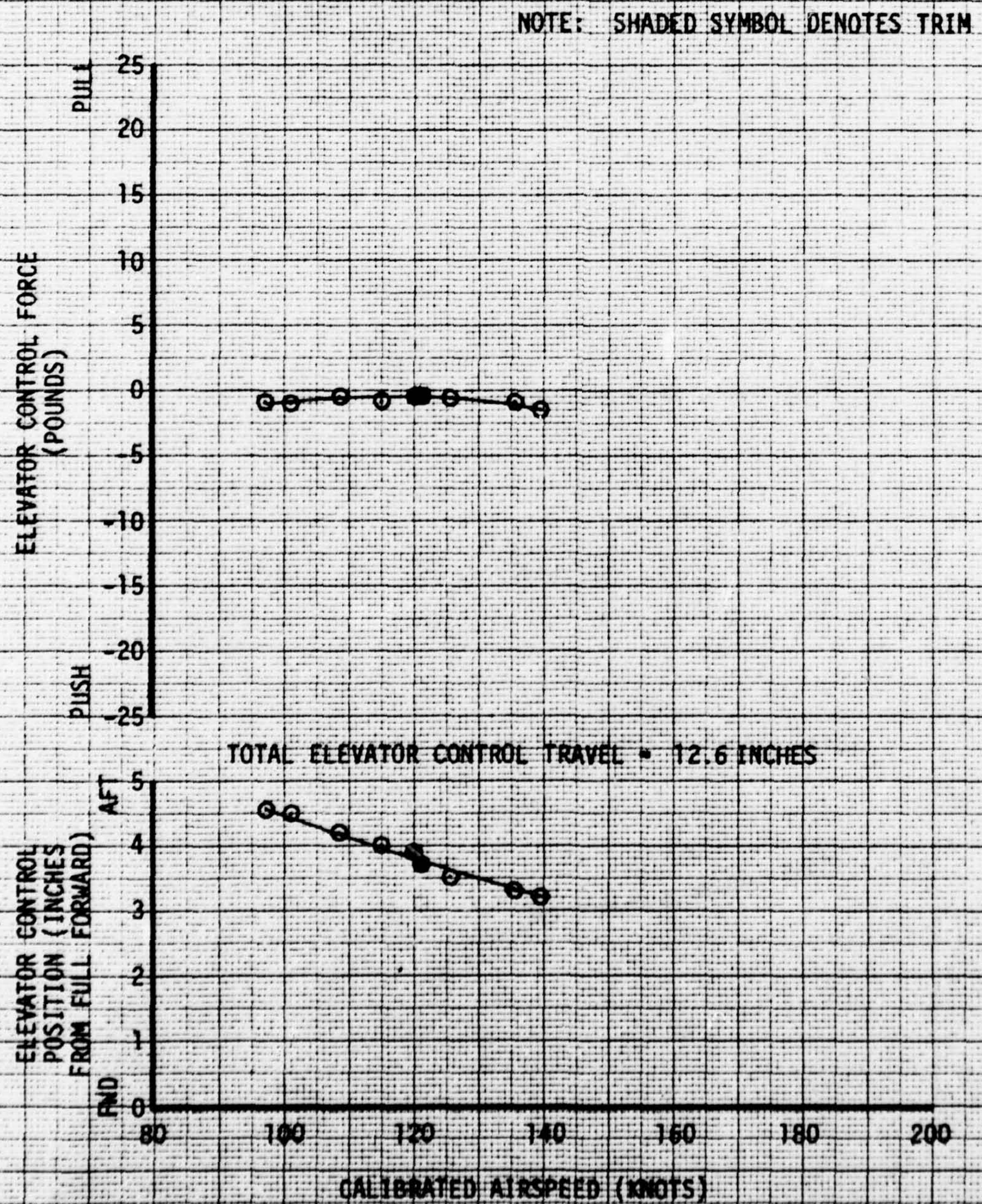


FIGURE 16
STATIC LATERAL-DIRECTIONAL STABILITY
TS-2A USFS S/N N404DF

AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
18780	215.9	7600	16.4		2300	CLEAN	LEVEL FLT

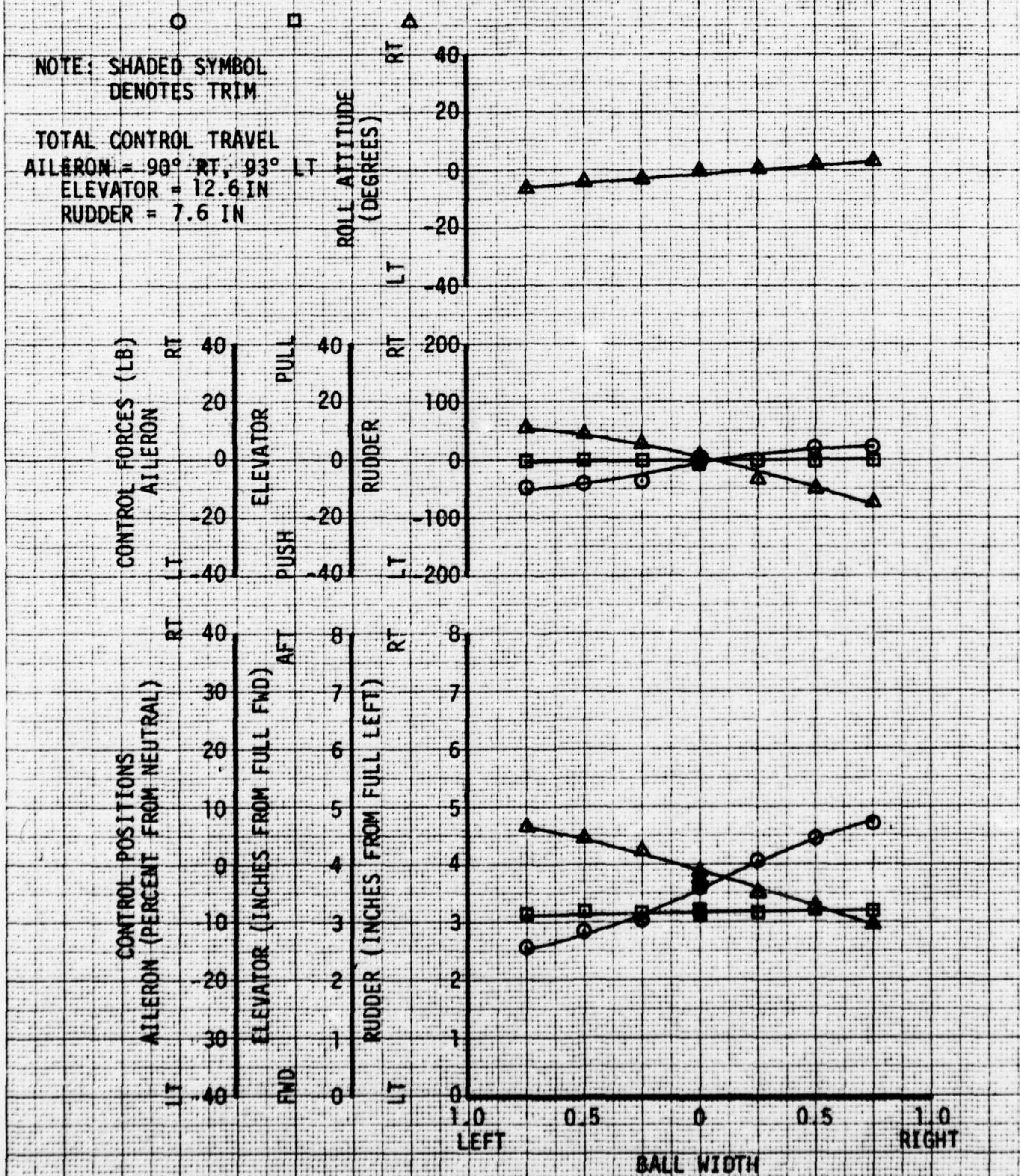


FIGURE 17
STATIC LATERAL-DIRECTIONAL STABILITY
TS-2A USFS S/N N404DF

AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
25860	215.8	7720	18.5	120	2300	CLEAN	LEVEL FLT
○	□	△					

NOTE: SHADED SYMBOL DENOTES TRIM

TOTAL CONTROL TRAVEL
AILERON = 90° RT, 93° LT
ELEVATOR = 12.6 IN
RUDDER = 7.6 IN

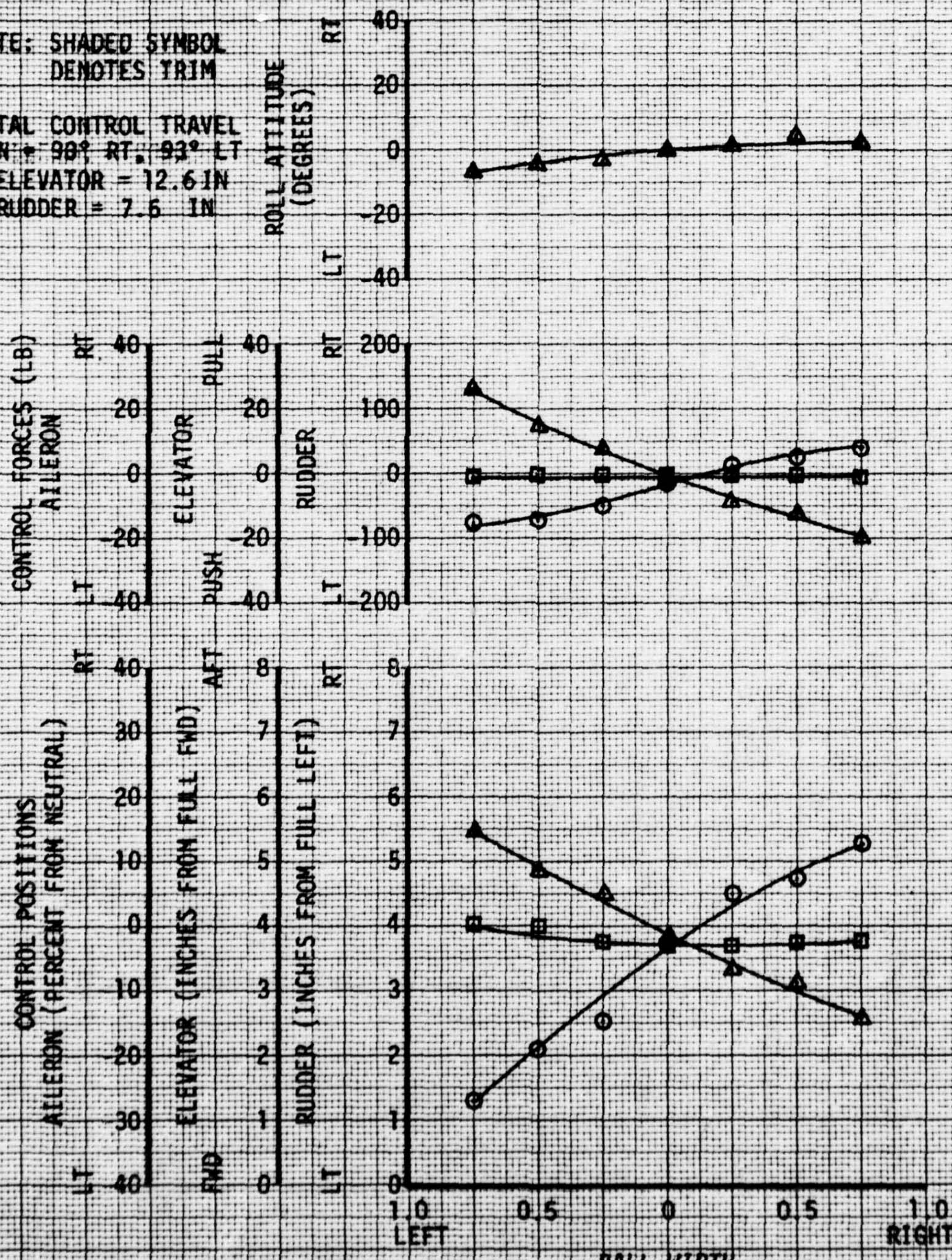


FIGURE 18
MANEUVERING STABILITY
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CALIBRATED AIRSPEED	FLT CONDITION
○	18655	216.0	8560	12.5	2300	140	WIND UP TURN
□	18790	216.0	8700	9.0	2300	120	WIND UP TURN

NOTE: AIRCRAFT CONFIGURATION IS CLEAN
SHADED SYMBOLS DENOTE LEFT TURN

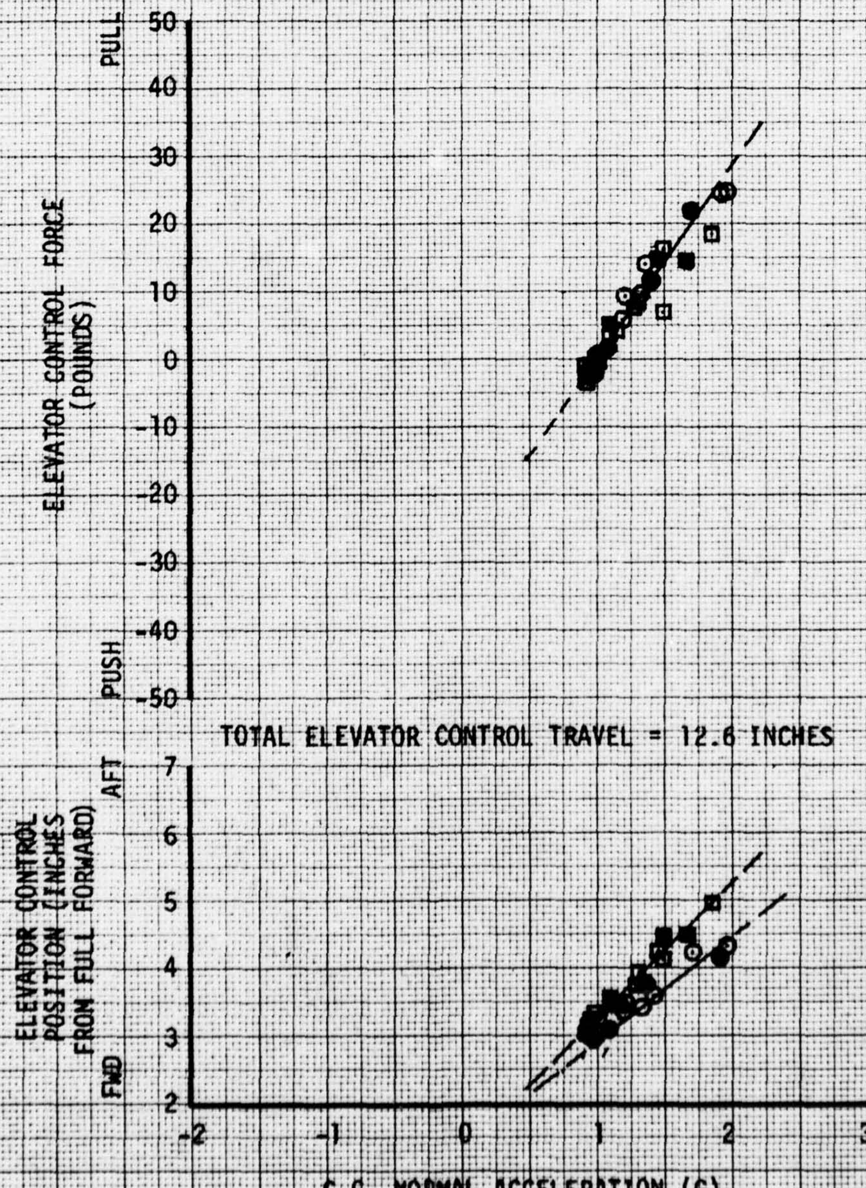


FIGURE 19
MANEUVERING STABILITY
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CALIBRATED AIRSPEED	FLT CONDITION
○	26130	215.7	7080	12.0	2300	140	WIND UP TURN
□	26030	215.7	7160	11.5	2300	120	WIND UP TURN

NOTE: AIRCRAFT CONFIGURATION IS CLEAN
SHADED SYMBOLS DENOTE LEFT TURN

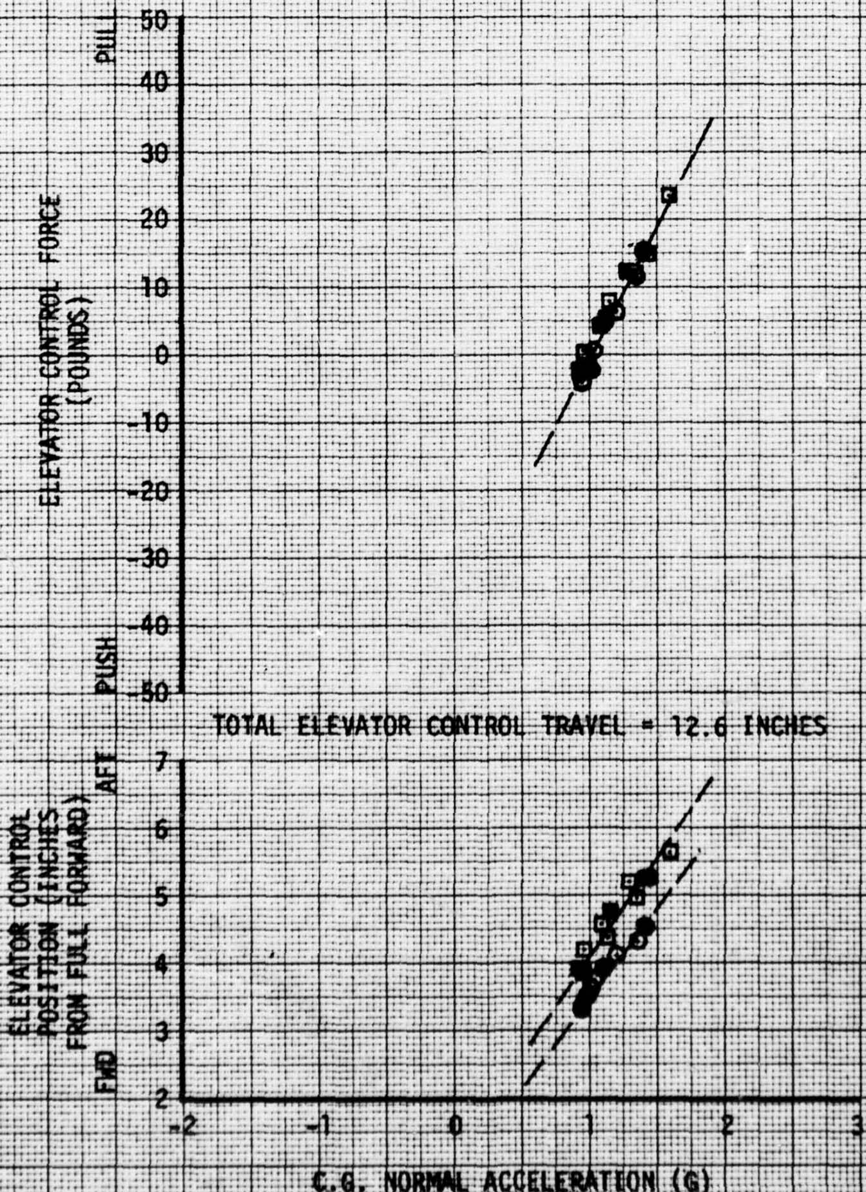


FIGURE 20
MANEUVERING STABILITY
T5-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CALIBRATED AIRSPEED	FLT CONDITION
○	18530	216.0	7520	14.0	2300	140	PULL UP AND PUSH OVER
□	18410	216.1	7660	13.5	2300	120	PULL UP AND PUSH OVER

NOTE: AIRCRAFT CONFIGURATION IS CLEAN

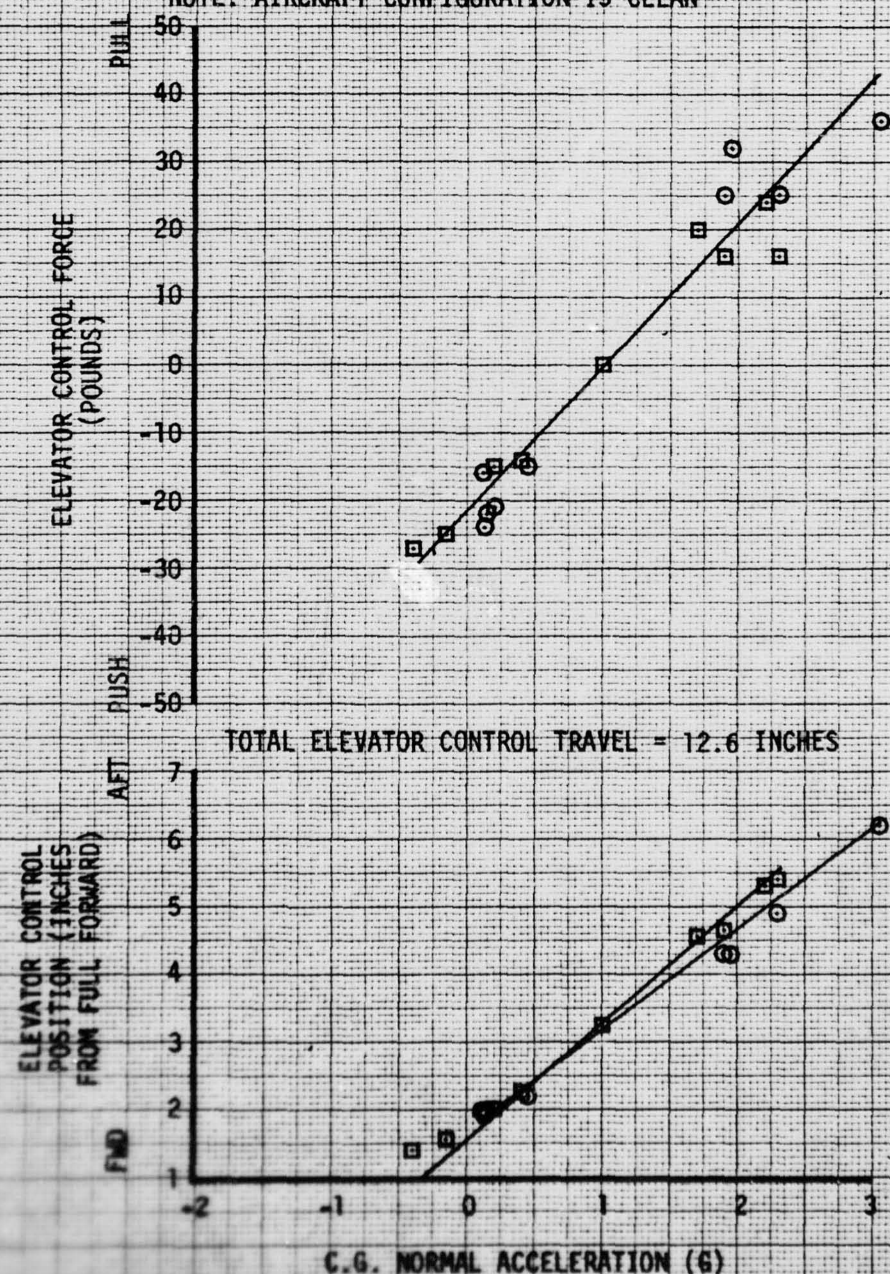


FIGURE 21
MANEUVERING STABILITY
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CALIBRATED AIRSPEED	FLT CONDITION
○	26130	215.7	7280	11.5	2300	140	PULL UP AND PUSH OVER
□	25880	215.8	7140	11.5	2300	120	PULL UP AND PUSH OVER

NOTE: AIRCRAFT CONFIGURATION IS CLEAN

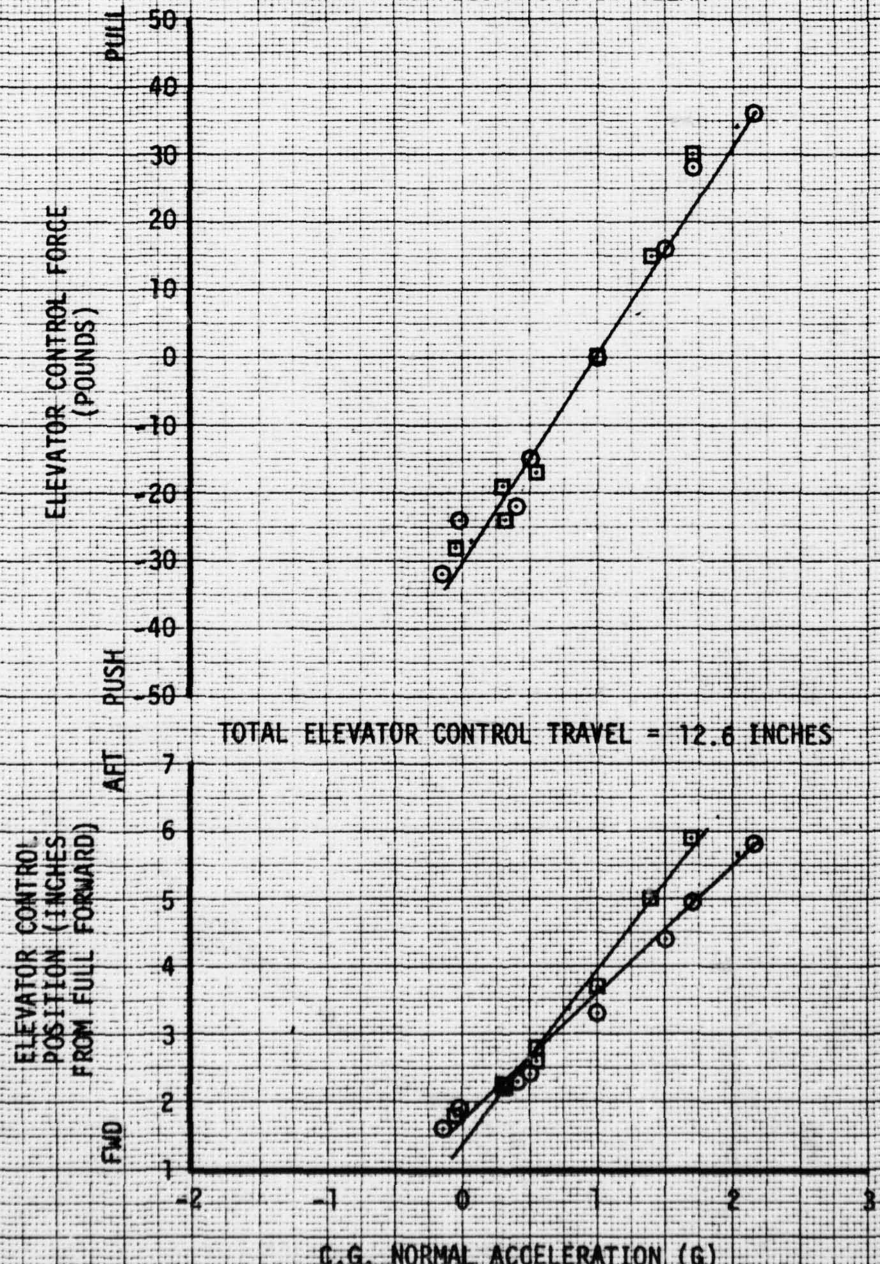


FIGURE 22
ROLL PERFORMANCE
TS-2A USFS S/N N404DF

	GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
SOLID LINE / SHORT DASH	25760	216.0	7500	16.5	2300	122	CLEAN	LEVEL FLIGHT

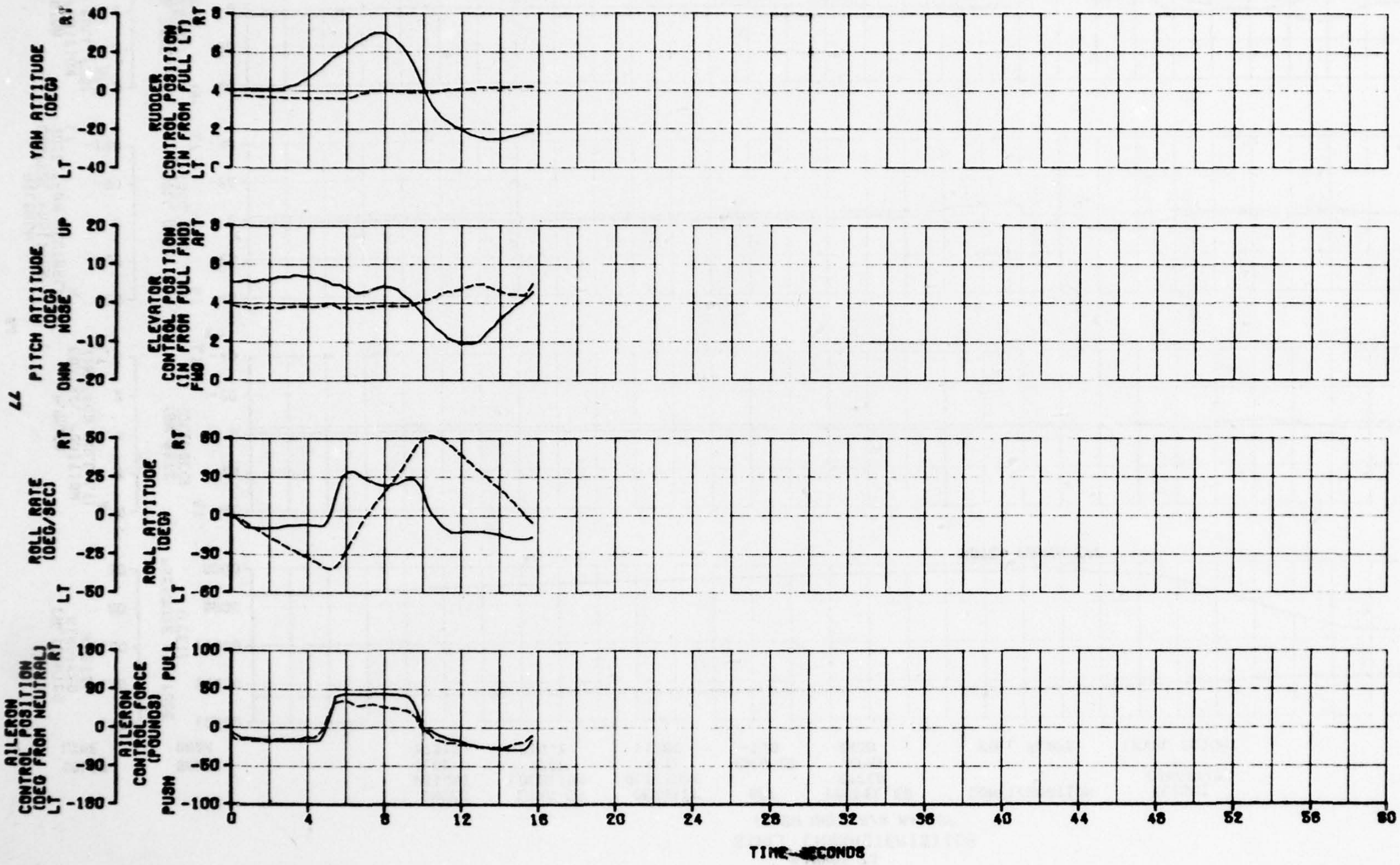


FIGURE 23
STALL CHARACTERISTICS
TS-2A USFS S/N N404DF

SOLID LINE	SHORT DASH	GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
		17310	216.2	11680	-1.0	2400	FULL FLAPS	LEVEL FLIGHT

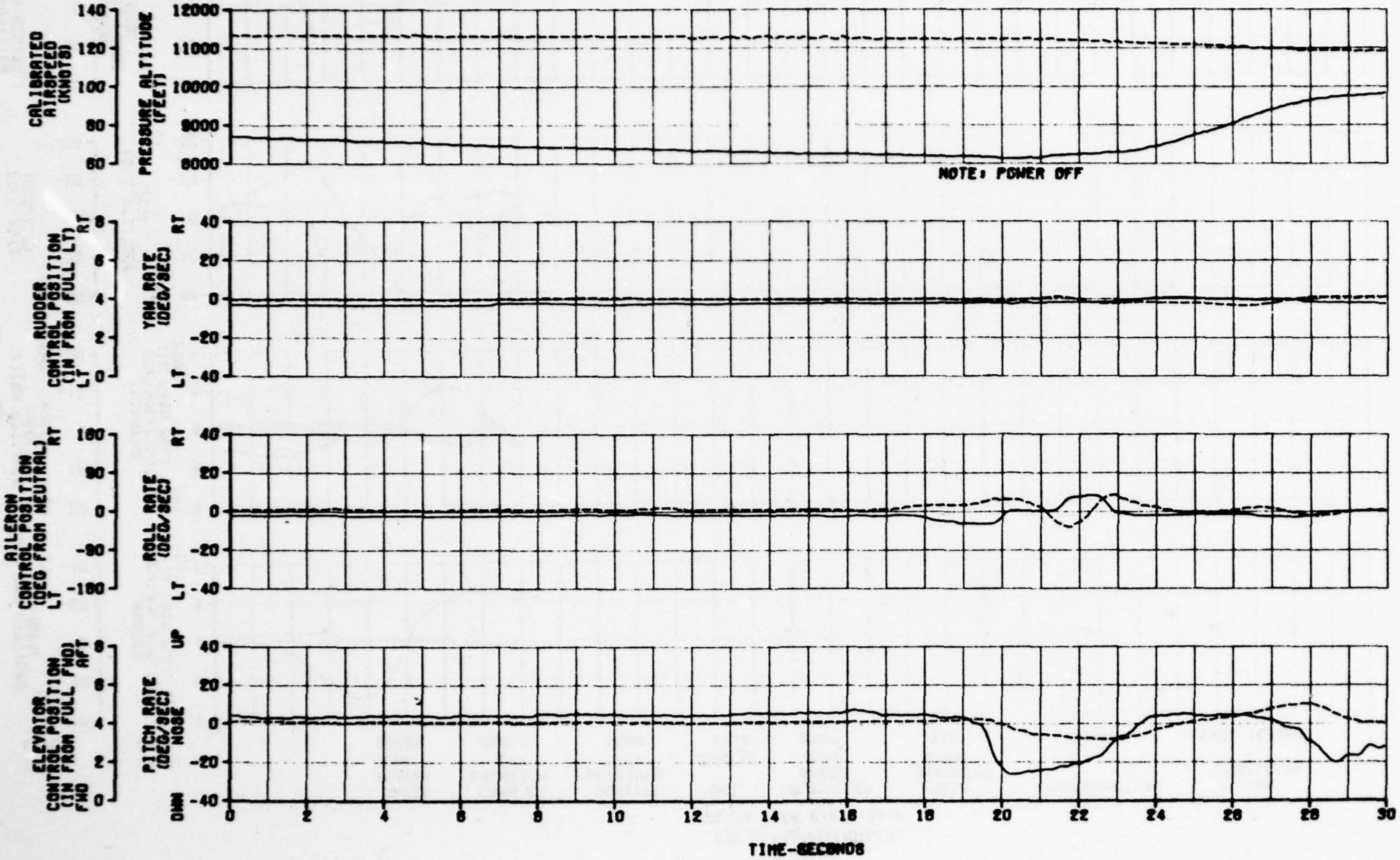


FIGURE 24
STALL CHARACTERISTICS
T6-2A USFS S/N N404DF

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
17900	216.4	11380	2.5	2400	FULL FLAPS	LEVEL FLIGHT

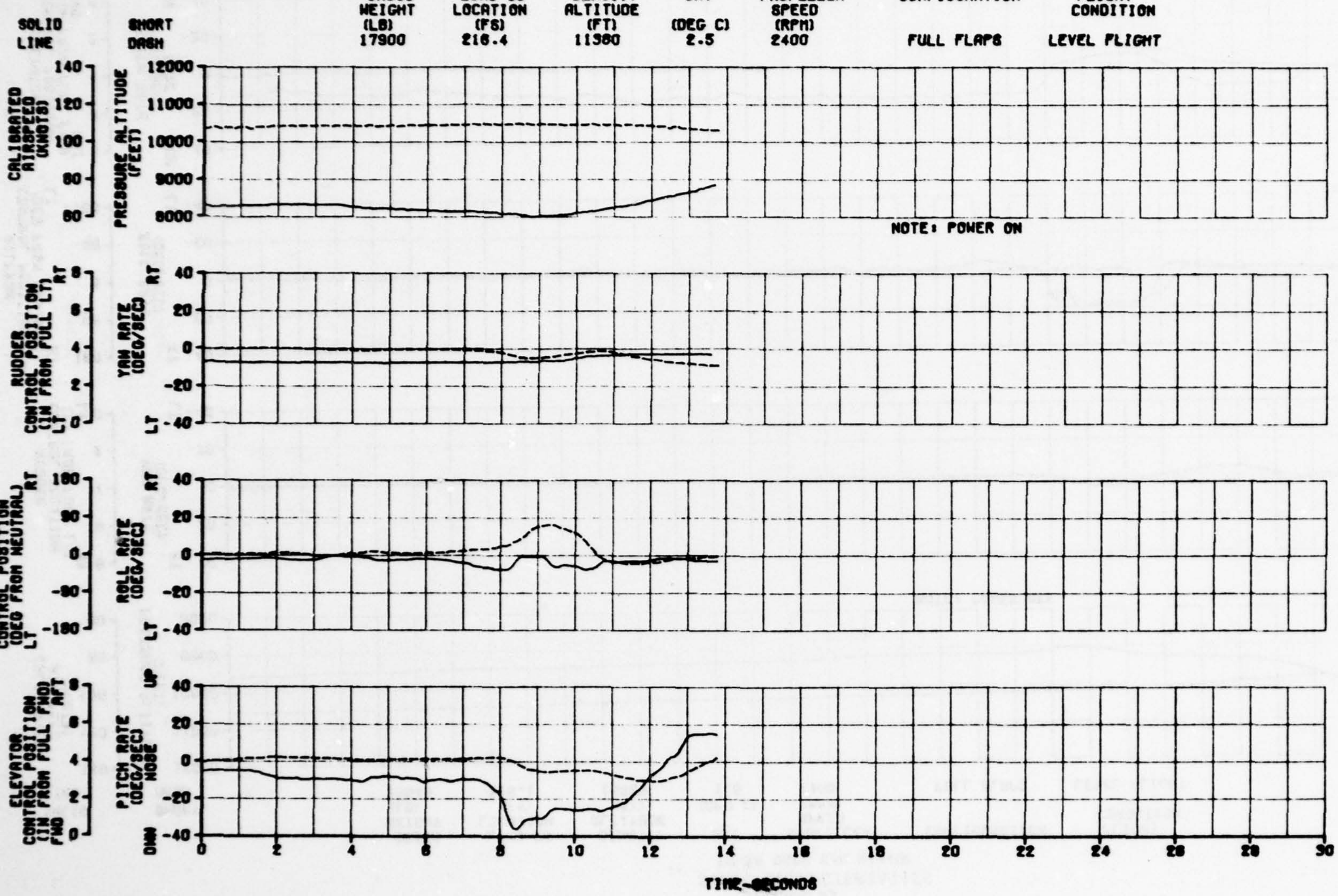


FIGURE 25
STALL CHARACTERISTICS
TS-2A USFS S/N N404DF

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
25540	216.1	10920	1.5	2400	FULL FLAPS	LEVEL FLIGHT

SOLID LINE — SHORT DASH

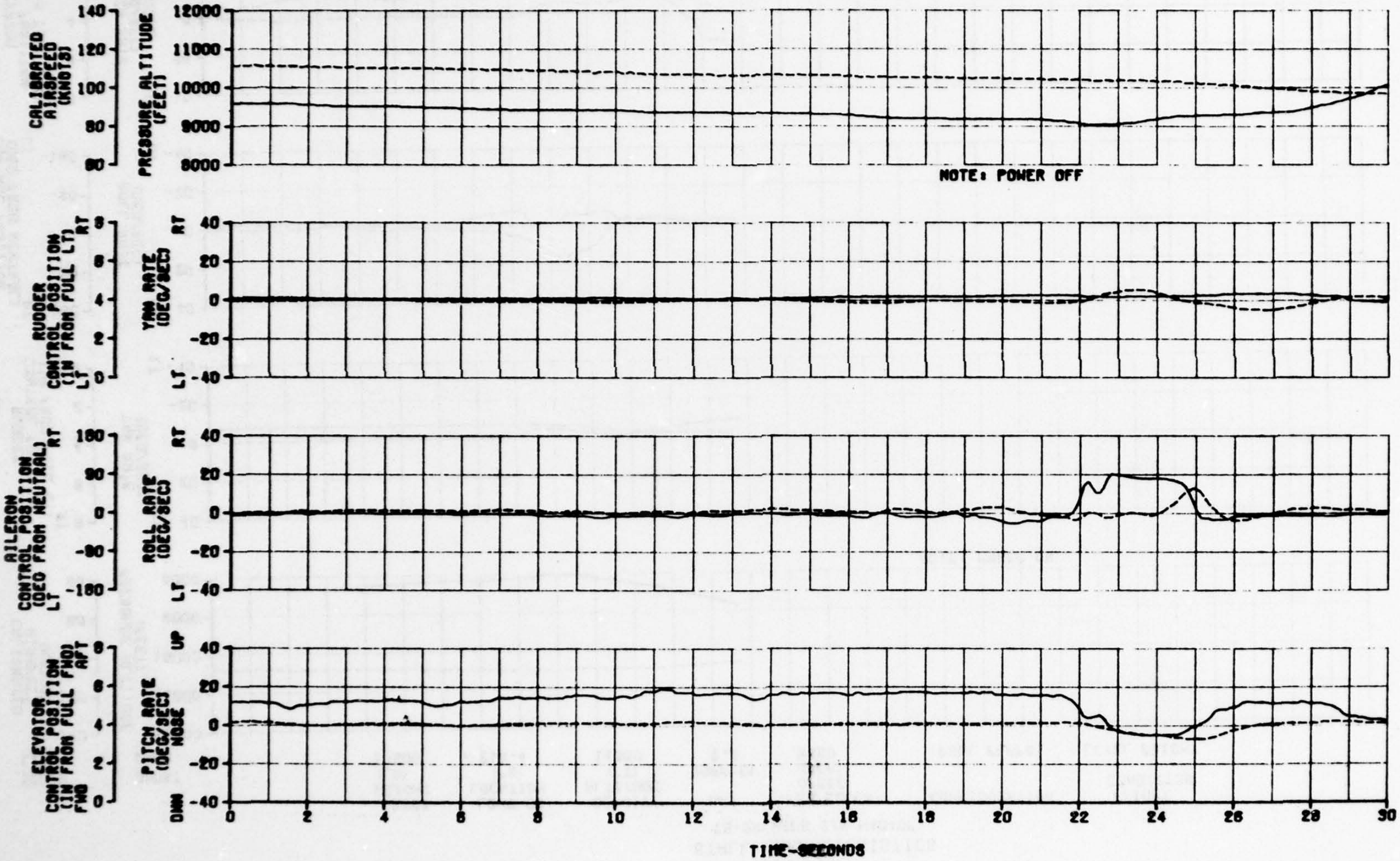


FIGURE 26
STALL CHARACTERISTICS
TS-2A USPS S/N N404DF

SOLID LINE	SHORT DASH	GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
		17900	216.4	11320	2.0	2400	FULL FLAPS	LEVEL FLIGHT

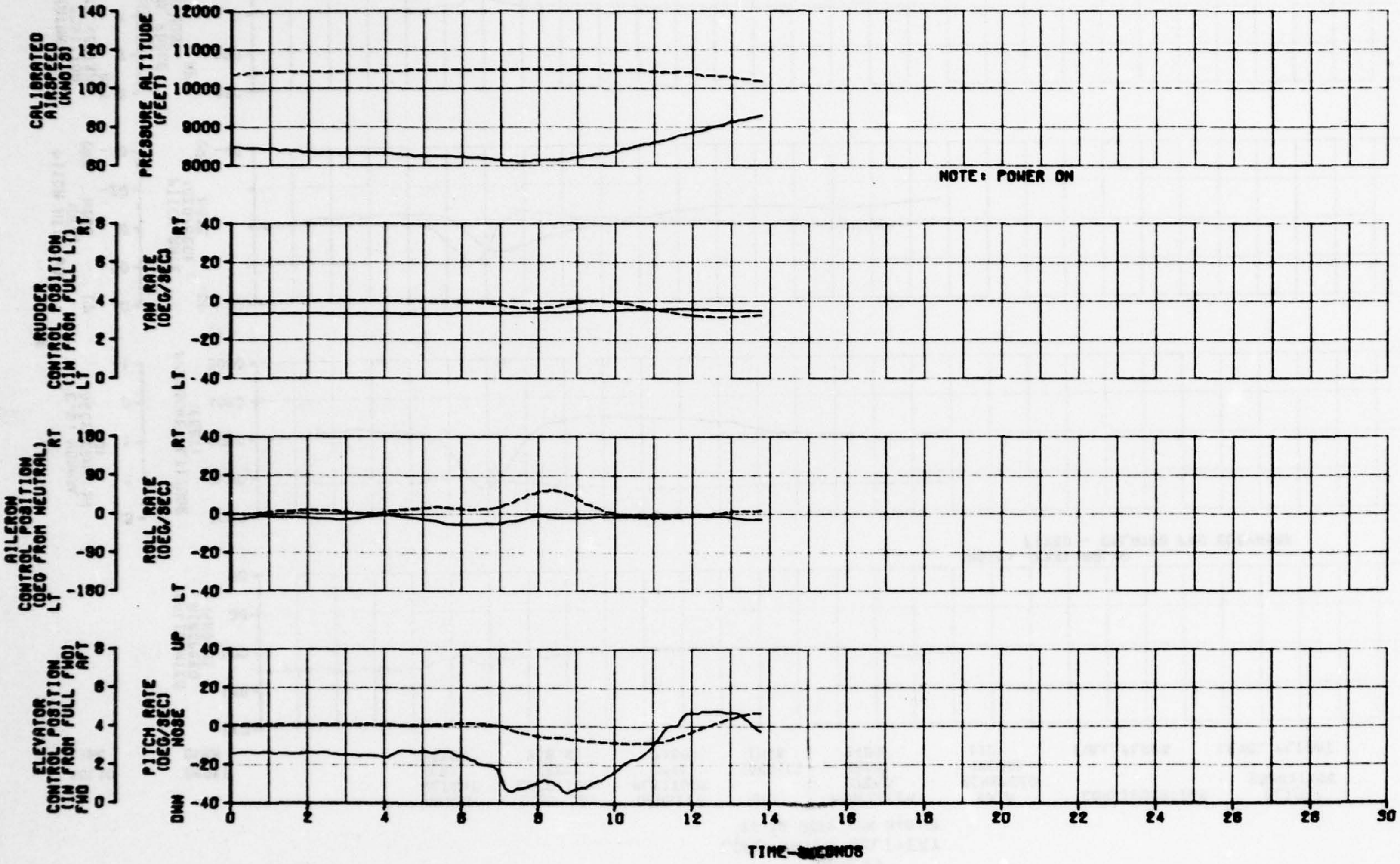


FIGURE 27
LOAD WATER DELIVERY
T6-2A USFS S/N N404DF

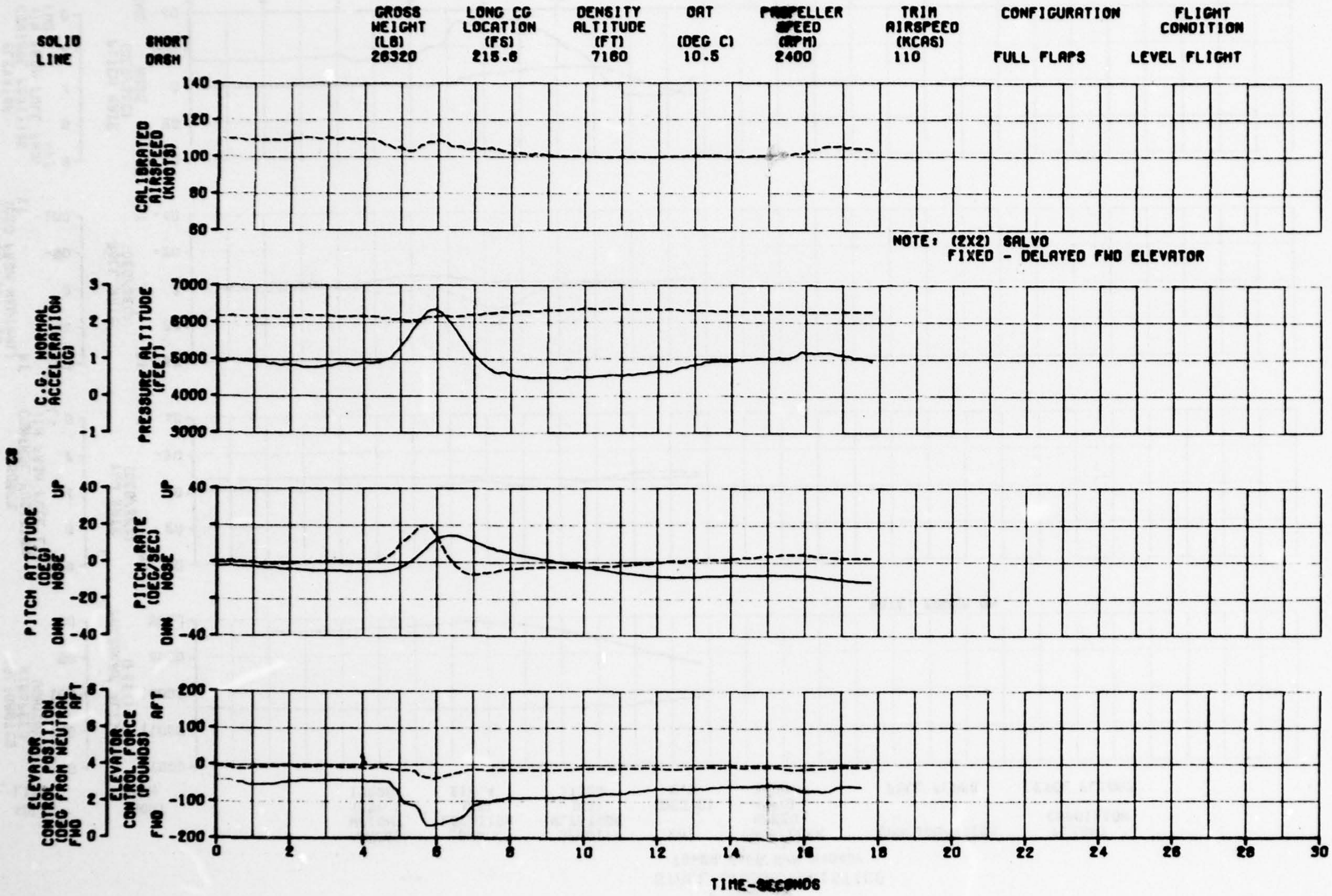


FIGURE 28
LOAD WATER DELIVERY
T6-2A USFS S/N N404DF

		GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
SOLID LINE	SHORT DASH	26320	215.6	7040	10.0	2400	130	FULL FLAPS	LEVEL FLIGHT

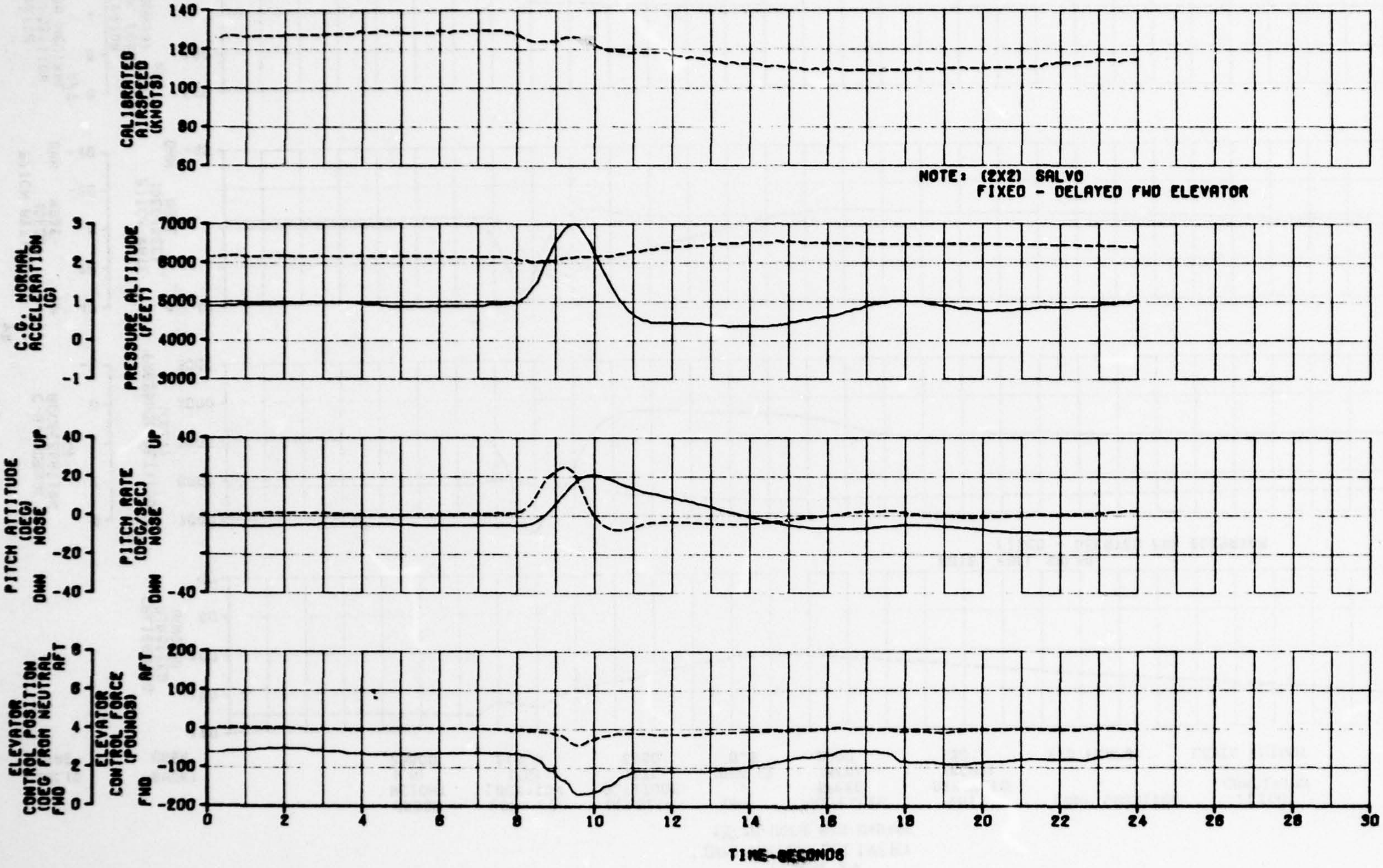


FIGURE 29
LOAD WATER DELIVERY
TS-2A USFS S/N N404OF

		GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
SOLID LINE	SHORT DASH	26800	215.8	6660	8.5	2400	130	2/3 FLAPS	LEVEL FLIGHT

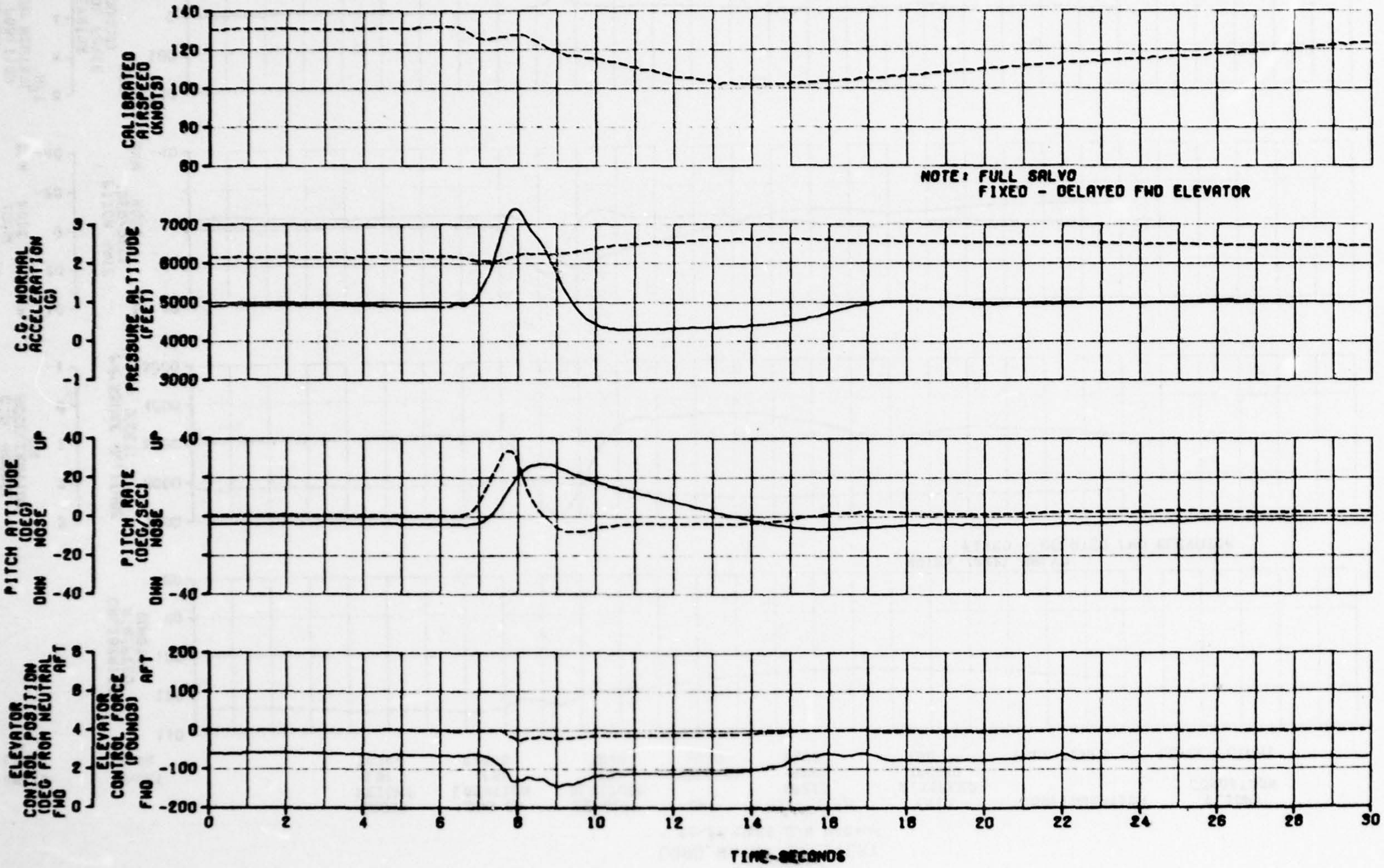


FIGURE 30
LOAD WATER DELIVERY
TS-2A USFS S/N N404DF

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
26320	215.6	7040	10.0	2400	130	2/3 FLAPS	LEVEL FLIGHT

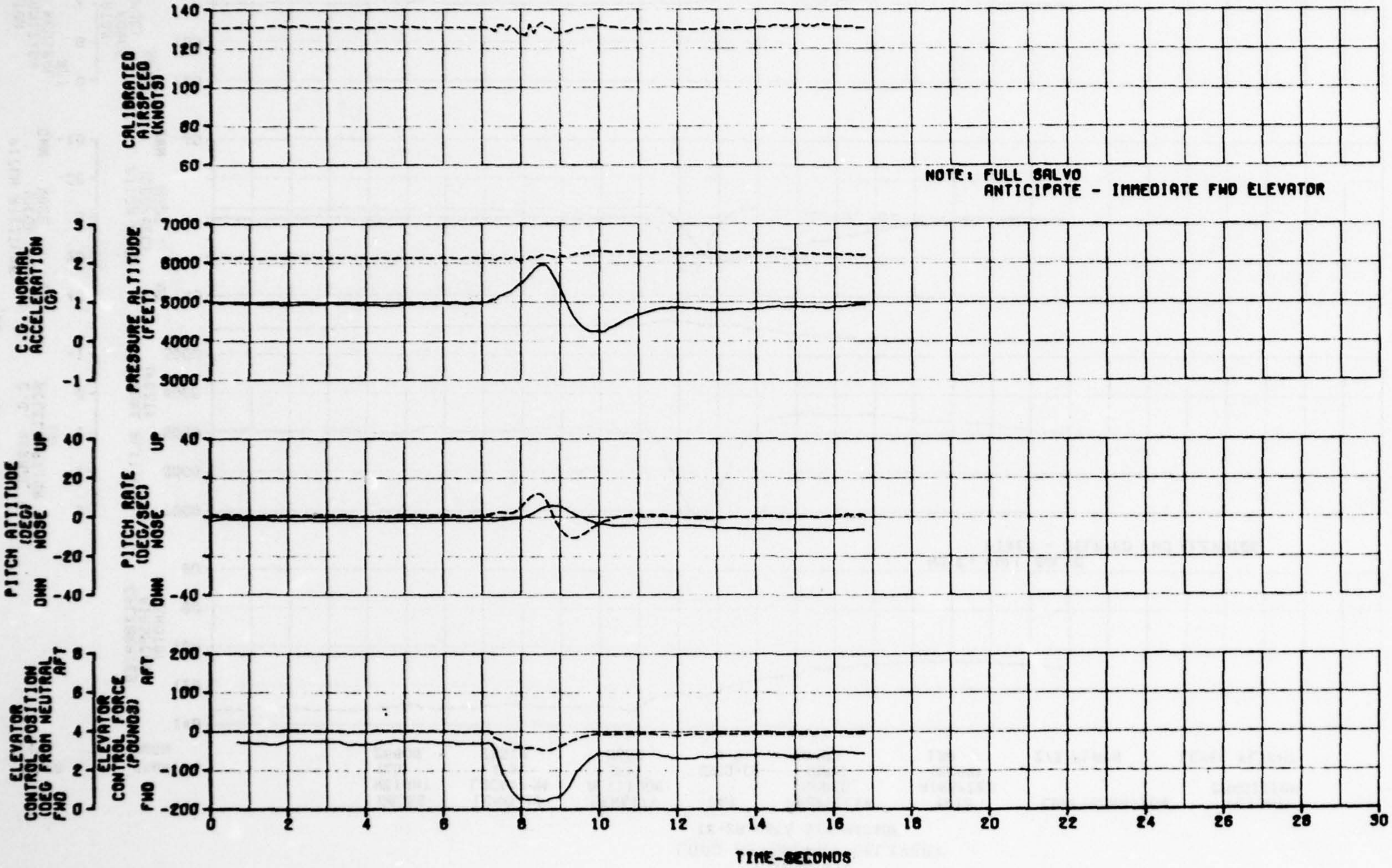


FIGURE 31
LOAD RETARDANT DELIVERY
TS-2A USFS S/N N404DF

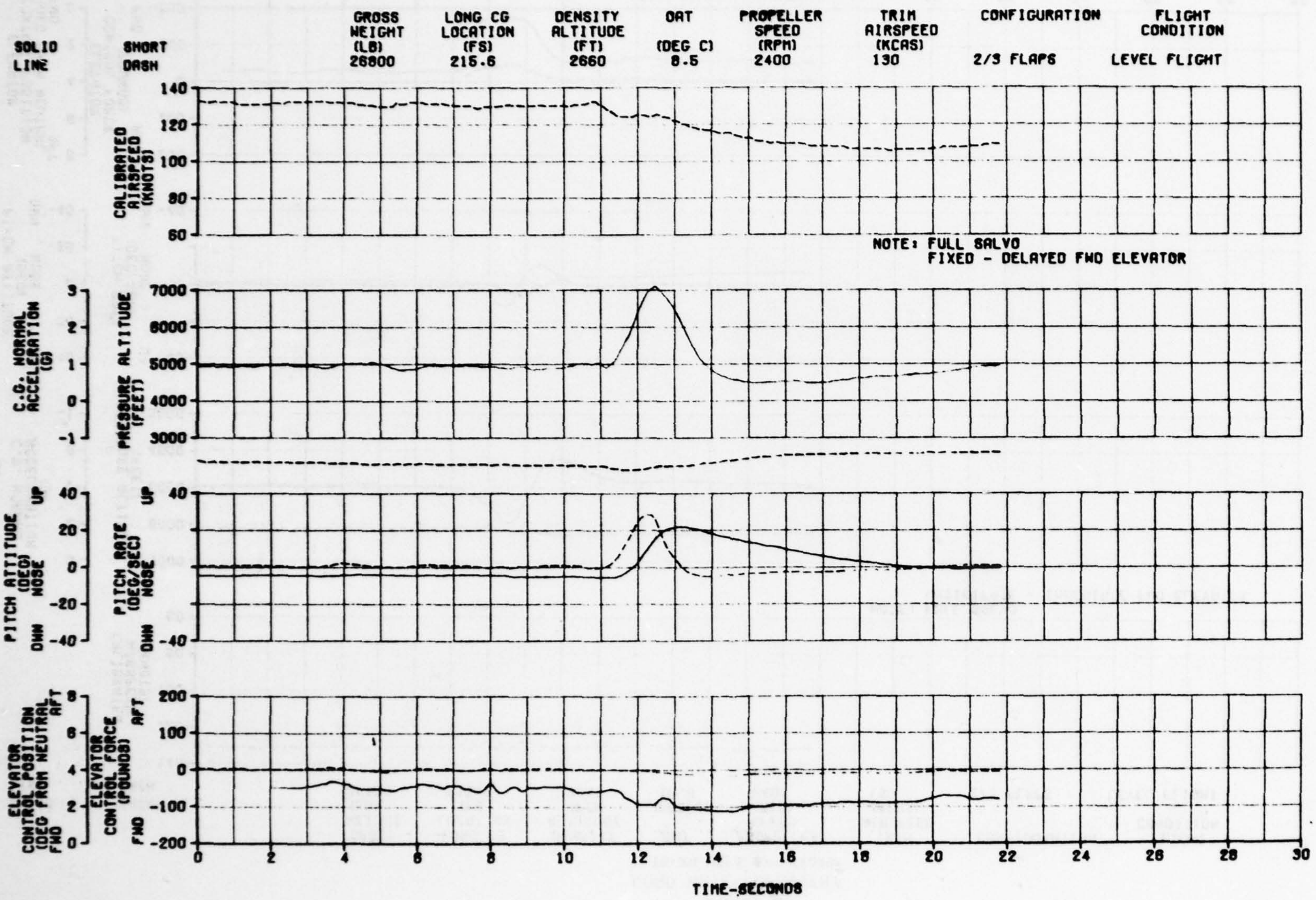


FIGURE 32
LOAD RETARDANT DELIVERY
TS-2A USFS S/N N404DF

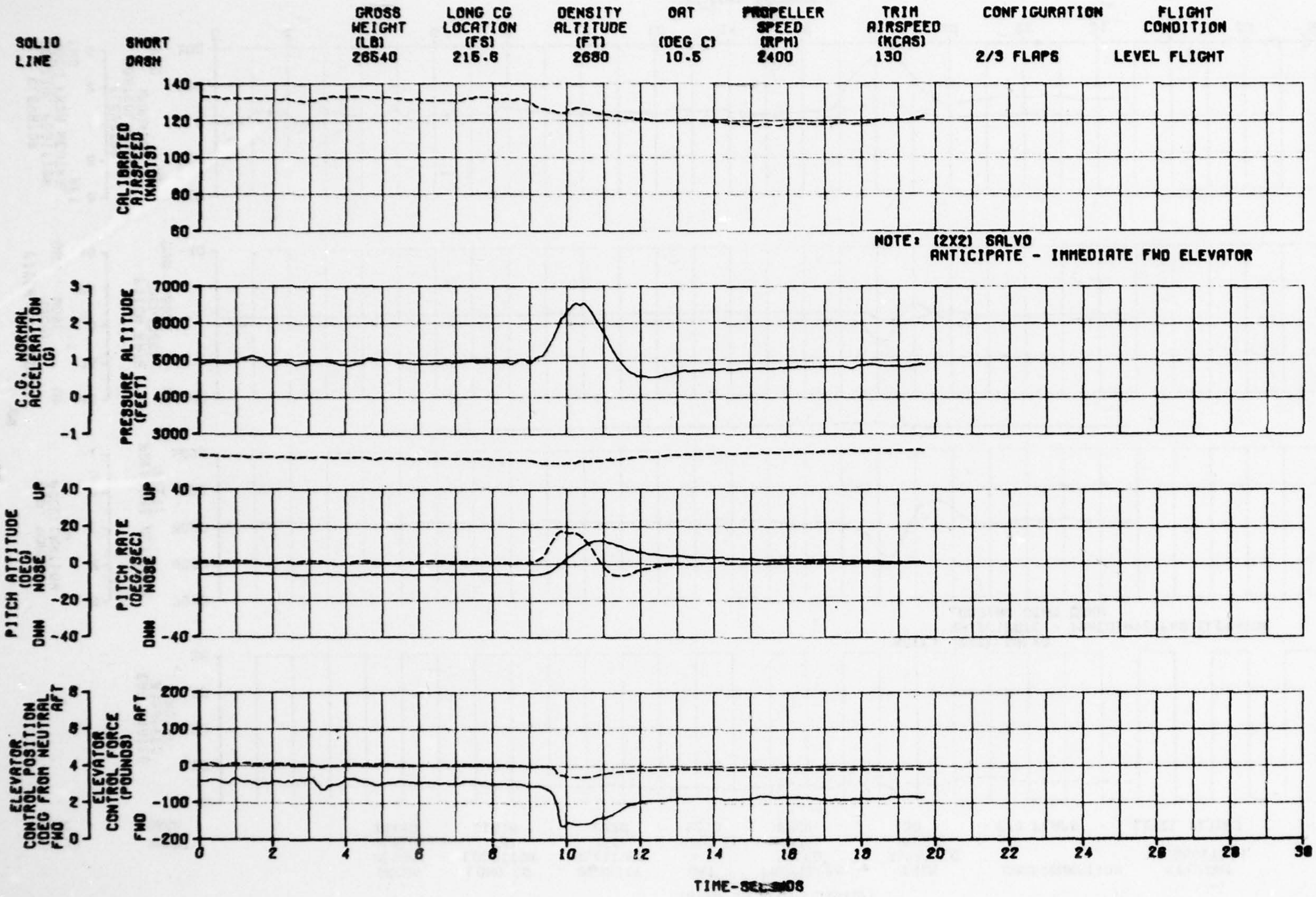
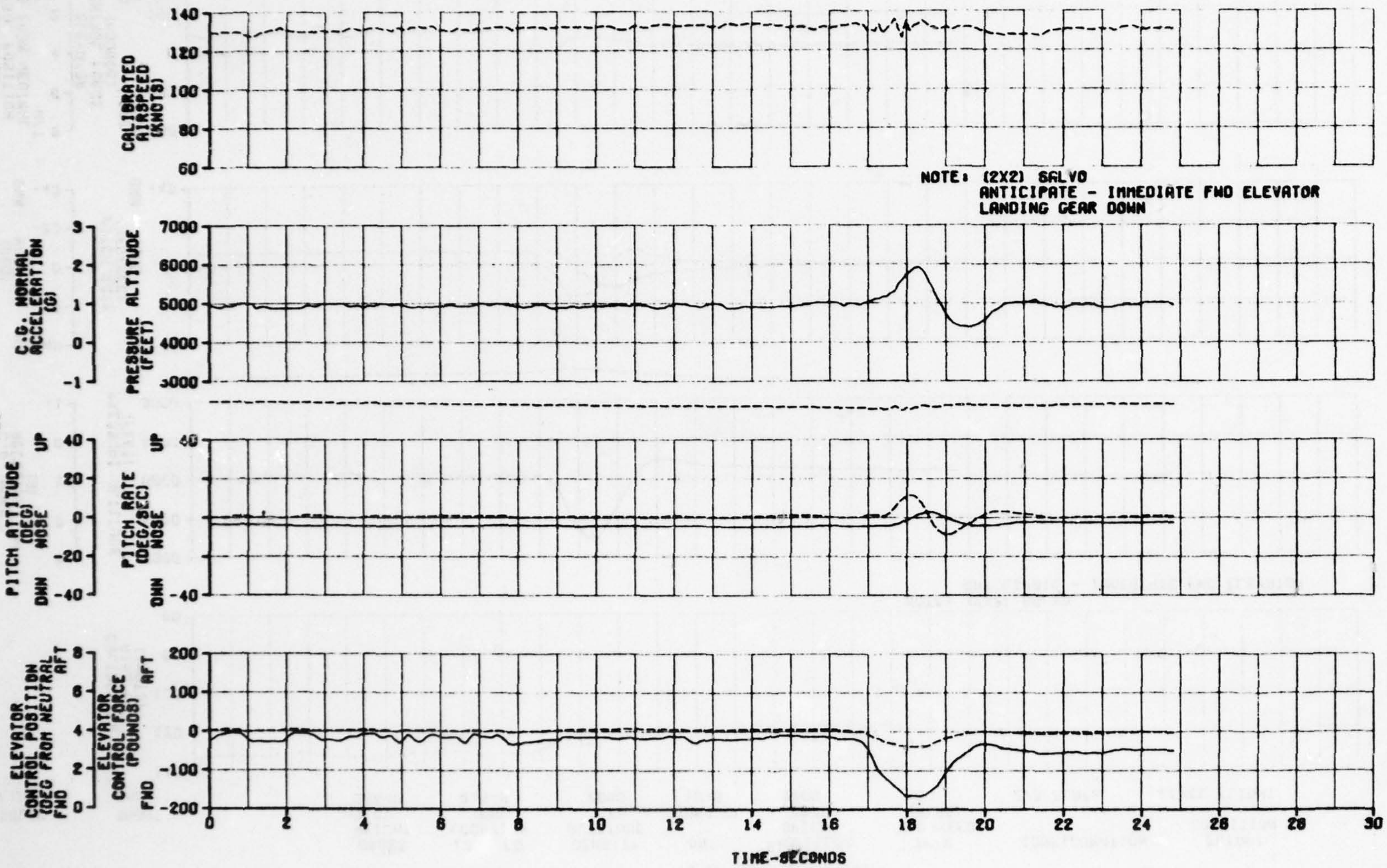


FIGURE 33
LOAD RETARDANT DELIVERY
T6-2A USFS S/N N404DF

		GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (DEG C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
SOLID LINE	SHORT DASH	26440	215.6	2720	12.0	2400	130	2/3 FLAPS	LEVEL FLIGHT



AD-A060 940 ARMY AVIATION ENGINEERING FLIGHT ACTIVITY EDWARDS AF--ETC F/G 1/3

TS-2A AIRTANKER EVALUATION. PHASE II.(U)

MAR 78 J C WATTS, E E BAILES, S C SPRING

UNCLASSIFIED USAAEFA-77-07 NL

2 OF 3

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DATE FILMED

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CONT

2 OF 3

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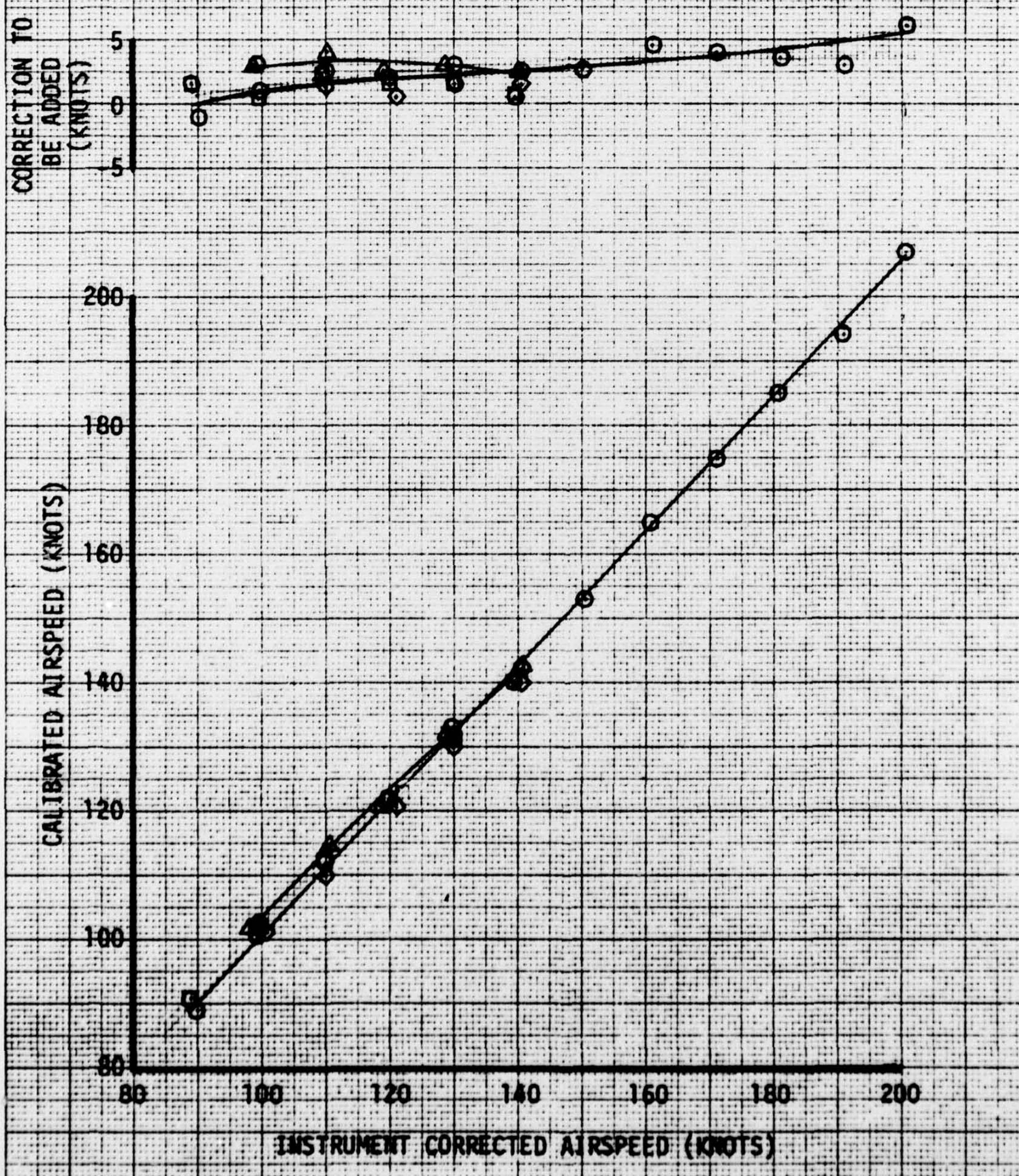
1.0 4.5 2.8 2.5
5.0 3.2 2.2
5.6 3.6
6.3 4.0 2.0
7.1
1.1 1.8
1.25 1.4 1.6

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

FIGURE 34
AIRSPEED CALIBRATION
TS-2A USFS S/N N404DF

SYM	AVG GROSS WEIGHT (LB)	AVG LONG C.G. LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	PROPELLER SPEED (RPM)	CONFIG	FLT CONDITION
○	18290	216.4	7560	12.5	2300	CLEAN	LEVEL FLT
□	18290	216.4	7560	12.5	2300	FULL FLAPS	LEVEL FLT
△	18090	216.4	7520	10.3	2300	FULL FLAPS	DESCENT
◇	18090	216.4	7480	10.6	2300	CLEAN	CLIMB



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April 1979

USAAEFA Project No. 77-07

Final Report

TS-2A Airtanker Evaluation
Phase II

United States Army Aviation Engineering Flight Activity
Edwards Air Force Base, California 93523

Page 47, paragraph 1

Change:

$$\Delta BHP_s = BHP_t - BHP_s$$

To-

$$BHP_s = BHP_t + \Delta BHP$$

To symbol list below equation 1, add-

BHP_s = Brake horsepower (standard)

BHP_t = Brake horsepower (test)

F_{p_s} = Mean net propeller thrust (standard)

F_{p_t} = Mean net propeller thrust (test)

V_t = Test true airspeed (ft/sec)

Change:

$$F_p = \Delta BHP/V_{t(avg)}$$

All changes have been incorporated. Previous Errata Sheets for this report are obsolete and should be destroyed.

To-

$$F_{p_s} = (BHP_s/V_{t(avg)})550$$

$$F_{p_t} = (BHP_t/V_{t(avg)})550$$

Page 48, paragraph 2

Add to symbol list-

σ_t = Test density ratio

Page 48, paragraph 3

Change equation 4 to read:

$$Sg_{s1} = \frac{Sg}{1 + \frac{2g\ Sg \sin\theta}{V_{TO}^2}}$$

Change Sg_{SL} in symbol list to Sg_{s1}

Page 49, equation 5

Remove parentheses in equation 5, to read:

$$\left(\frac{W_t}{W_s} F_{p_s} - F_{p_t}\right)$$

Change the symbol list below equation 6:

From θ_t to σ_t

From θ_s to σ_s

Remove-

LBF/FT = Pound Force

Page 50, paragraph 4

Change symbol list below equation 8:

Units of aircraft energy level to (ft-lb)

Page 51, paragraph 6

Equation 10 should read:

$$R/C_{corr} = R/C_{tapeline} + \Delta R/C_1 + \Delta R/C_2$$

Equation 11 should read:

$$\frac{dh_p}{dt} \; [(T_t + 273.15)/(T_s + 273.15)]^{\frac{1}{2}}$$

Change the following in symbol list below equation 11:

Remove-

dh/dt = Recorded R/C

Add-

h_p = Pressure altitude

Page 52

Remove symbol list below equation 12

Add to symbol list below equation 13

W_t = Test gross weight

Page 54, paragraph 13

Change (θ) to (σ)